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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT
7 RUE ANCELLE, 92200 NEUILLY-SUR-SEINE, FRANCE

AGARD LECTURE SERIES 206

Aging Combat Aircraft Fleets — Long Term Applications

(les Conséquences à long terme du vieillissement des flottes d'avions de combat)

The material in this publication was assembled to support a Lecture Series under the sponsorship of the Structures and Materials Panel of AGARD and the Consultant and Exchange Programme of AGARD presented on 7-8 October 1996 in Madrid, Spain, 10-11 October 1996 in Pomezia, Italy, 4-5 November 1996 in Atlanta, USA and 22-24 January 1997 in Brussels, Belgium.

NORTH ATLANTIC TREATY ORGANIZATION

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North Atlantic Treaty Organization
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— Continuously stimulating advances in the aerospace sciences relevant to strengthening the common defence posture;

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Aging Combat Aircraft Fleets — Long Term Implications

(AGARD LS-206)

Executive Summary

Because national defence budgets are being reduced, NATO Air forces will have to continue to operate existing fleets for many years beyond what was anticipated several years ago.

This lecture series will provide technical information to fleet operators, managers, industry personnel responsible for upgrading the capabilities of combat aircraft, maintenance personnel at air logistics centres, and specialists involved with design. It will assist them in making tactical adjustments to better manage aging fleets and to be able to deal with aging-related problems, as they arise, in order to maintain the existing fleets at their maximum operational capacity and air superiority in a changing environment.

The aspect of retrofit/rejuvenation of aging aircraft will be highlighted through presentations relating to three front-line combat aircraft in the inventories of NATO air forces.

Les conséquences à long terme du vieillissement des flottes d'avions de combat

(AGARD LS-206)

Synthèse

La réduction des budgets nationaux de défense et par voie de conséquence des forces aériennes de l'OTAN conduiront à maintenir en service les flottes existantes bien au delà des prévisions faites il y a quelques années.

Cette série de conférences fournira des informations techniques aux experts chargé de la mise en œuvre des flottes aériennes, aux managers, aux industriels chargés de l'actualisation des performances des avions de combat, de la maintenance, aux spécialistes des centres de logistique aériens ainsi qu'aux responsables de la conception des aéronefs au sein de l'OTAN. Elle leur permettra de faire les ajustements techniques nécessaires pour mieux gérer le vieillissement des flottes et de mieux surmonter les problèmes associés, au moment où ils surviennent, pour maintenir aux flottes en ligne leur capacité opérationnelle maximum et leur supériorité dans un environnement changeant.

Le problème de la reconfiguration/rajeunissement sera illustré par des présentations sur les enseignement tirés de la remise à niveau des trois des principaux avions de combat en première ligne dans les forces aériennes de l'OTAN.

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Aging Combat Aircraft Fleets—Long Term Implications
Introduction to Lecture Series

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Aging Aircraft concerns have dramatically escalated in military and civilian quarters alike during the past few years. The percentage of aircraft that are being operated beyond their design lives is ever increasing. As of 1993, approximately 51% of the aircraft in the U.S. Air Force (USAF) inventory were over 15-years old and 44% were over 20-years old. Yet, some aircraft models that have already served NATO for 30 years or more may need to be retained for another two decades. Due to NATO's changing role, which includes peace keeping missions remote from home bases, the requirement of unimpaired high operational capacity, higher utilization of its air fleets, and budgetary constraints, prospects are for aging aircraft problems to continue to become more acute.

Aging Aircraft has several connotations. Among them: (a) technological obsolescence, (b) the need for system upgrading, (c) changing mission requirements unanticipated during design specification and development, (d) the specter of runaway maintenance costs, (e) decreased safety, (f) impairment of fleet readiness, and (g) the unavailability of home depot facilities. If there is one common denominator among the various connotations it is the cost of operating aging aircraft. This Lecture Series (LS) will be relevant to all aforementioned items, but the theme, "how to minimize the cost of operating aging aircraft fleets while maintaining their effectiveness" will be a recurring one.

The purpose of this LS is to promote awareness among the user community about technical solutions which can ameliorate some of the concerns. Thus, the LS has been designed to provide technical information to the fleet operators and managers within NATO to assist them in making tactical adjustments in managing aging aircraft fleets more economically and be able to deal with aging related problems as they arise. The LS will also be of value to industry personnel responsible for upgrading the capabilities of combat aircraft, maintenance personnel at air logistics centers, and specialists involved with design of repairs and prescription of inspection methods.

The LS has been designed to cover systems update and structural airworthiness aspects of aging, fixed-wing aircraft. Due to the immediacy of aging aircraft problems being confronted by NATO fleet managers, the lectures have been structured to stress "what problems have been encountered in aging aircraft," "how have they been dealt with," and "what lessons have been learned."

The aspect of maintaining aging aircraft cannot be overstated as Figure 1 [1] shows. It may be seen that for the EF-111A aircraft, on an average, the man-hours required for scheduled inspection and repair of each aircraft in the depot has risen from about 2200 hours in 1985 to about 8000 hours today. The figure also shows how much more expensive the cost of maintaining the airframe structure of an aging aircraft is relative to other systems. In 1985, structure-related maintenance accounted for some 20% of the total man-hours, but today that figure has risen to almost 50%.

The substantial cost savings that can be realized through application of advanced technologies is vividly demonstrated in the example of the T-38 aircraft [2]. The originally designed 66% spar of the T-38 wing has been cracking at around 2500 flight hours when the aircraft is flown in a lead-in fighter role. Two options are available for extending the life of the fleet till the year 2015. One of the options involves replacement and maintenance of the spar as per the original design; the second involves use of a more effective design that involves a different material selection. Assuming the fleet size as 490 aircraft, Table 1 shows a comparison of estimated costs and projected cost savings of the order of 40% by using the advanced technology solution. Such a cost advantage is principally due to the advanced technology solution requiring only one replacement campaign whereas the solution based on the original design will require that the spar be replaced twice. The support requirements for both options are projected to be the same, meaning that no new labor skills will be entailed by the advanced technology option. Thus, newer technology can offer enhanced fleet readiness as a collateral benefit.

Table 1. Cost Analysis of 66% T-38 SPAR

Cost Category	Current Solution	Advanced Technology Solution
1. Base and Intermediate Maintenance Costs	$ 353,531	$ 352,531
2. Depot Maintenance Costs (Repair)	$ 7,200,060	$ 3,600,030
3. Depot Maintenance Costs (Replacement)		
a. Material Cost	$34,805,680	$22,399,860
b. Labor Costs	$30,105,600	$16,016,140
Total Depot Maintenance Cost	$64,911,280	$38,416,000
TOTAL SUPPORT COSTS (1+2+3)	$72,464,871	$42,369,561
SUPPORT COST SAVINGS		$30,095,310
SUPPORT COST % SAVINGS		41.5%

Corrosion is the bane of aging aircraft structures and accounts for a large fraction of maintenance hours and cost. Reference 3 provides a striking example of the benefit of inserting newer technologies to combat corrosion related costs associated with aging aircraft. Corrosion related data that were gathered earlier were analyzed [3]. The study had a two-fold purpose:

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(a) to determine if the data support the assertion that new technology and techniques for manufacturing and maintaining aircraft will reduce corrosion, and hence maintenance costs, and (b) to attempt to forecast the life-cycle cost (LCC) of corrosion maintenance of a weapon system.

The technology insertion measurement, as described in [3], was performed on the actual field and depot data collected for fleets of aircraft over a two-year period starting in 1989. Thus, it was possible to compare the cumulative corrosion-related costs on a per-aircraft basis for the C-5A fleet versus those for the C-5B fleet which included technology insertion. Similar cost comparison analysis for the C-130E and C-130H fleets were performed. Table 2 shows the cost comparisons and the tremendous savings that can accrue through upgrades involving newer material, new manufacturing technology, and improved maintenance practices.

Table 2. Cumulative Corrosion-Related Repair Cost (Same Age and Corrosion Severity Exposure) (Per Aircraft)

Aircraft	Base Level	Base and Depot Level	Percent Improvement, %
C-5A C-5B	$299,810 $244,762	$2,326,175 $1,007,417	56.69
C-130E C-130H	$ 83,023 $ 38,771	$3,947,503 $1,270,379	67.82

Reference 3 also provides the results of LCC calculations that were made, using the Modular Life Cycle Cost Model (MLCCM) owned by the USAF's Wright Laboratory, to gage the benefit of technology insertion (see Table 3). As can be seen in the table, the ratio of potential savings in corrosion maintenance costs through technology insertion (C-5A versus C-5B fleets, and the C-130E versus C-130H fleets) is remarkable.

Table 3. MLCCM Results, Per Aircraft

Aircraft	Result
C-5A to C-5B	$ 806,620
C-130E to C-130H	$3,079,241

Aspects relating to performance upgrade, mid-life updates, life extension, and retrofit of three front line combat aircraft will be covered in separate presentations, highlighting solutions that have actually been implemented. The four types of aircraft—the F-16, the Tornado, the CF-18 and the C-135 aircraft—are important constituents in NATO's inventory. A presentation about the life extension efforts concerning the C/KC-135 aircraft has been included in the LS because of its tactical role and relationship to combat aircraft fleets. The four case studies will describe implementation strategies and discuss ways to improve the ability of an airframe to accommodate new systems to meet present-day mission requirements.

The problem of aging aircraft has highlighted the need for research and development to resolve certain unique technical issues such as widespread fatigue damage. To address the issues, the USAF has been conducting research that pertains to several aging aircraft systems. The latter include airframe structure, propulsion, avionics, and subsystems such as cockpit controls and displays and hydraulic actuators for flight controls and landing gear. In the spirit of "what solutions are available," the LS will include a presentation about the research program.

Technical solutions pertaining to maintenance management and pro-active rehabilitation and retrofit schemes will also be addressed in a presentation, titled "Repair/Refurbishment of Military Aircraft." The paper will also discuss some retrofit schemes that utilize advanced materials.

A presentation about flight loads and monitoring has been included in the LS because combat aircraft are increasingly being called upon to serve new roles, as in Bosnia—roles that were not envisaged even a few years ago. Some such missions require new stores configurations which can result in expansion of the flight envelope, causing increases in consumption of fatigue life. Thus, it is important to know the load profiles associated with the missions because by affecting the stress and fatigue life they can hasten the aging process.

A presentation concerning the F-16 corrosion prevention program within USAF will highlight the importance to be accorded to corrosion prevention and control. A video of the material that is used for instructing the Federal Aviation Administration's inspector work force about various aspects of corrosion will also be shown.

Aircraft structures are designed and built to withstand the cyclic loads that are encountered during service. As they age, degradation of their structural strength due to fatigue is inevitable, thus requiring strict management of the integrity of the structure. The management scheme will also need to account for possible synergistic effects of fatigue and battle damage or damage due to other external sources. The most troubling aspect of fatigue, which is particular to aging aircraft, is the phenomenon of widespread fatigue damage (WFD). Onset of WFD usually causes a precipitous fall in structural strength. The USAF uses the damage tolerance philosophy, embodied in their Airframe Structural Integrity Program (ASIP) and their Engine Structural Integrity Program (ENSIP), to ensure continued airworthiness of its fleet. A presentation on fatigue and damage tolerance will describe the methods used for structural integrity management.

Inspection is a cornerstone of the damage tolerance philosophy, because through inspection, cracks, corrosion, and accidental damage are required to be detected prior to structural failure. Many systems are available to detect damage and more systems and equipment are in various stages of development. However, an inspection method that is suitable for inspecting a certain structural part may be unsuited to a different inspection task. The LS includes a presentation to highlight the relative advantages of various inspection methods.

Inspection reliability is another aspect of inspection and is covered in a presentation on that subject. Inspection reliability data are essential to deriving inspection thresholds and inspection intervals—elements of every maintenance program for the constituents within a fleet. Frequency and the method of inspection are primary drivers of maintenance costs and thus life-cycle costs. On the other hand, structural safety also depends on inspection reliability, i.e., the ability to detect damage in a timely fashion.

Automated systems for monitoring the health of aircraft structures are in various stages of development. Such systems have the potential to reduce maintenance costs. The cost benefits are primarily realized through reduction in inspection requirements and sustaining support. The prospects of implementing health monitoring for Force Management will be discussed in a presentation on the subject.

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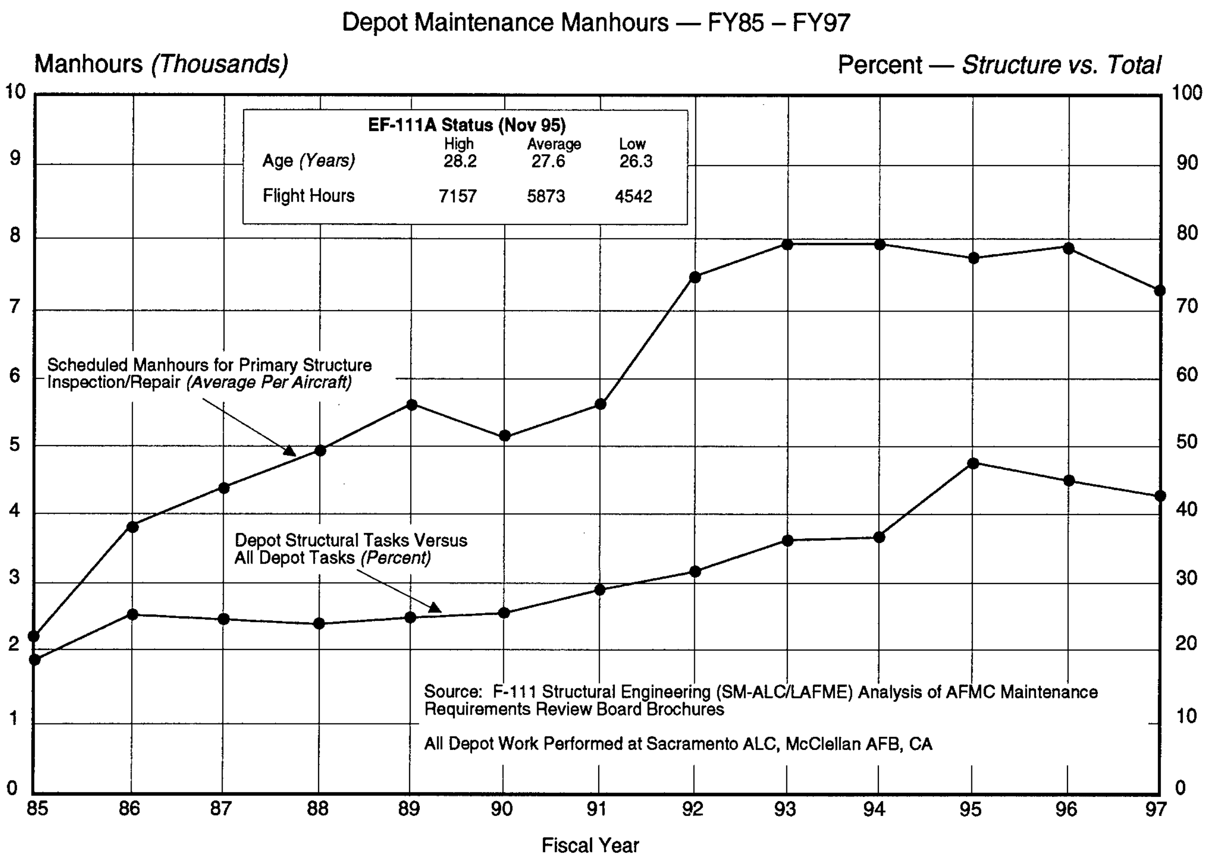

Figure 1. EF-111A Structural Inspection History

USAF AGING AIRCRAFT PROGRAM

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SUMMARY

The United States Air Force (USAF) has numerous aircraft that have already met or exceeded their original design service lives; many of these aircraft will considerably exceed their original life goals before they are retired from the inventory. Technologies are needed which will extend the lives and/or reduce the costs of these aging aircraft. Such technologies will ensure flight safety, reduce maintenance and repair requirements and their associated costs, and increase operational readiness. A description of the USAF aging aircraft systems research and development efforts that are being conducted in the technology categories of Structural Integrity, Nondestructive Evaluation/Inspection, Avionics, Propulsion and Subsystems is presented. The structural integrity area considers damage that can degrade the service lives of aging aircraft; the technologies required to ensure aircraft structural integrity with such damage present are identified. Typical types of damage considered include corrosion, fatigue cracking, and the potential interaction of corrosion and fatigue. Also, the possible occurrence of widespread fatigue damage (WFD), which becomes more likely as aircraft structure ages, is addressed. Life extension methodology is described that includes the development of analytical and experimental procedures for the repair (e.g., composite patches) of metallic structure that will restore or extend the full service life of the damaged structure. The detrimental effects of severe dynamic loads and environments on structural integrity is considered (e.g., buffeting, limit cycle oscillations, and acoustics). A structural health monitoring capability is described that includes on-board sensors which detect corrosion, fatigue cracking, and severe environments for assessing their detrimental effects on the structural integrity of aging aircraft. Nondestructive Evaluation/Inspection technologies are described for detecting corrosion and fatigue cracks (e.g., multiple-site damage). Avionics emphasis is given to electronic parts obsolescence, computational obsolescence, and integrated modular avionics. Propulsion technologies described will reduce the risk of turbine engine failures and provide cost savings by utilizing JP-8+100 fuel, maintenance-free batteries and thin dense chrome bearings. Subsystems technology developments described include electric actuators that replace hydraulic actuators for flight control, smart diagnostics for flight control, more durable and cost effective transparencies, more durable landing gear components, longer life tires, and more modernized and standardized cockpit control/displays.

1. IMPORTANCE OF AGING AIRCRAFT RESEARCH

The United States Air Force (USAF) has an aging aircraft fleet, as illustrated in Figs. 1 and 2. These figures present three-dimensional plots of selected aircraft, the number of these aircraft, and their age. The plots are based on 1994 data. Several of the larger USAF aircraft (e.g., bombers, transports and tankers) are described in Fig. 1. For illustrative purposes, consider the C/KC-135 aircraft. There are a large number of these aircraft that are currently 30-40 years old. One scenario is to keep these aircraft operational until the year 2040. If this occurs, these aircraft will have become 75-85 years old. This would indeed represent an aging aircraft fleet. Similar scenarios exist for the other aircraft depicted in Fig 1. Many of these aircraft have already met or exceeded their original design service lives; many of them will considerably exceed their original life goals before they are removed from the inventory.

Similarly, several of the smaller aircraft (e.g., fighters, attacks, and trainers) in the USAF inventory are described in Fig. 2. Many of these aircraft (e.g., F-4, F-111, T-37 and T-38)

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are approaching or have already reached 25 years in age. Like the larger aircraft, many of these smaller aircraft have already met or exceeded their original design service lives; many will considerably exceed them before they are removed from the inventory. Technologies are required to ensure that both large and small USAF aging aircraft can be operated safely and economically as they remain in the inventory. Such technologies and their development are described in this paper.

The importance of aging aircraft research can be further emphasized by one of the primary USAF customers for such research, the Air Mobility Command. In a letter dated 30 December 1994 to General Yates, Commander of the Air Force Materiel Command, it was indicated by General Rutherford, Commander of the Air Mobility Command, that he wanted to reemphasize the importance of aging aircraft research to his command. General Rutherford stated that the need for aging aircraft research, especially research dealing with corrosion, was a concern identified in the Air Mobility Master Plan for the past two years. His most immediate concern was for his oldest aircraft, the KC-135. He felt that the apparent key was to accomplish adequate research such that a severe problem could be predicted and effectively dealt with before it became a crisis.

General Yates responded to General Rutherford's comments via a letter dated 19 January 1995. General Yates indicated that he agreed with General Rutherford's emphasis on the importance of research and development to support the USAF aging aircraft fleet. General Yates stated that as part of the Fiscal Year 1996 (FY96) budgeting process, the science and technology community began an aging aircraft initiative which increased research in aging aircraft by over a third. He further indicated that Wright Laboratory recently formed an Aging Aircraft Customer Focus Integrated Product Team (CFIPT) which was working closely with the Oklahoma City Air Logistics Center (OC-ALC) and the Warner Robins Air Logistics Center (WR-ALC) to solve problems associated with the KC-135, C-141 and C-130 aging fleets.

2. CUSTOMERS

Primary customers for the USAF aging aircraft research and development technology results include the five Air Logistics Centers (ALCs): San Antonio (SA-ALC), Sacramento (SM-ALC), Ogden (OO-ALC), Oklahoma City (OC-ALC), and Warner Robins (WR-ALC). These ALCs are largely responsible for maintaining and keeping the USAF aging aircraft operationally ready. The System Program Offices (SPOs) for the different weapon systems, particularly those SPOs located at the ALCs, are also recipients of aging aircraft technologies. Other primary customers include the USAF Major Commands (MAJCOMs): Air Combat Command (ACC), Air Mobility Command (AMC), Air Education and Training Command (AETC), Air Force Special Operations Command (AFSOC), Air National Guard (ANG), and Air Force Reserve (AFRES). For example, the importance of aging aircraft research to the Air Mobility Command was described in the previous section. Finally, the aging aircraft technology results are also beneficial to the aircraft industry as a whole.

3. VISION

The vision of the USAF Aging Aircraft Systems Program is to develop and transition technologies to extend the lives and/or reduce the costs of aging aircraft. The successful achievement of this vision will enable the USAF to a) ensure the flight safety of its aging aircraft in order to avoid catastrophic failure, b) reduce maintenance and repair requirements and their associated costs, and c) increase the operational readiness of its aging aircraft, which could be critical in a wartime environment.

4. TECHNOLOGY CATEGORIES

The research efforts of the USAF Aging Aircraft Systems Program are grouped into five technology categories, as depicted in Fig. 3. These categories are Structural Integrity, Nondestructive Evaluation/Inspection, Avionics, Propulsion and Subsystems. A description of on-going and planned aging aircraft systems research and development in each of these categories is given in Sections 6 through 10 of this paper.

5. AFMC AGING AIRCRAFT WORKING GROUPS

AFMC Aging Aircraft Working Groups were established in January 1994 by the AFMC Aging Aircraft Steering Group, which consisted of Dr. Jim C.I. Chang (Air Force Office of Scientific Research, AFOSR), Mr. Otha Davenport (Engineering Directorate of Headquarters Air Force Materiel Command, HQ AFMC/EN) and Mr. O. Lester Smithers, Jr. (Wright Laboratory, WL). Since that time, Dr. Stephan Butler has replaced Mr. Otha Davenport as a steering group member. The overall purpose of the working groups is to assure the mission capability of the Air Force aging aircraft fleet. More specifically, the purpose of the working groups is to assure that there is effective communication between the technology developers (i.e., AFOSR and WL) and the technology users (i.e., five ALCs). The technology users identify aging aircraft issues of most concern to them. The technology developers ensure that these issues are addressed in their research, development, test and evaluation (RDT&E) plans. A product of the working groups is a set of aging aircraft RDT&E technology roadmaps that include basic research, exploratory development and advanced development technology programs.

AFMC Aging Aircraft Working Groups currently exist for two of the five technology categories previously identified: Structural Integrity and Nondestructive Evaluation/Inspection. Leadership for the Structural Integrity Working Group is provided by Mr. James Rudd (WL), Dr. Spencer Wu (AFOSR) and Mr. Daniel Register (WR-ALC). Similarly, leaders of the Nondestructive Evaluation/Inspection Working Group are Mr. Tobey Cordell (WL), Dr. Walter Jones (AFOSR) and Mr. Ralph Paglia (SA-ALC). These working groups meet at least twice a year, in conjunction with the Air Force Aging Aircraft Conference and the USAF Structural Integrity Program Conference. Working groups are currently being formed for the other three technologies categories: Avionics, Propulsion and Subsystems.

6. STRUCTURAL INTEGRITY

The Structural Integrity technology category consists of five sub-categories: Widespread Fatigue Damage (WFD), Corrosion/Fatigue, Repairs, Dynamics and Health Monitoring. Each of these sub-categories is described in the following paragraphs of this section.

6.1 Widespread Fatigue Damage (WFD)

One of the most significant forms of material and structural degradation for aging aircraft is widespread fatigue damage. Such damage was recently experienced by C-141 transport aircraft, as shown in Fig. 4. This damage occurred at wing station 405 (i.e., the chordwise splice of the inner and outer wing) and the fuel-transfer weep holes (i.e., lower surface, integrally-stiffened inner wing). Analysis methods are being developed for predicting when the onset of widespread fatigue damage will occur. This analytical capability will be used to ensure the integrity of structures with multiple-site and/or multiple-element damage present. The remaining structural life and residual strength for these damaged structures will be determined. The technology will be used to optimize the inspection intervals for detecting such damage. Also, it will provide the capability to assess the risk of catastrophic failure for aging aircraft with such damage present. Both deterministic and probabilistic approaches will be considered in the development of the analysis methods described above.

6.2 Corrosion/Fatigue

Another significant form of material and structural degradation is corrosion and its potential interactions with fatigue. Such damage is illustrated in Fig. 5. Corrosion is currently a major problem for the C/KC-135 transport and tanker aircraft. Analysis methods are being developed for predicting the detrimental effects on structural integrity of corrosion and its potential interactions with fatigue. It should be noted that corrosion sites are also potential sites for widespread fatigue damage. Hence, it may be necessary to couple the complex phenomenon of corrosion/fatigue with that of widespread fatigue damage that was described in the previous paragraph. Also, the presence of prior corrosion and/or a corrosive environment may result in accelerated fatigue crack growth rates and reduced structural lives. The analysis methods developed here will be used to ensure the integrity of structures that are pre-corroded and/or exposed to a corrosive environment. The capability will be provided for determining the remaining structural life and residual strength of such damaged structures. The capability to optimize inspection intervals for detecting such damage and harsh environments will also be provided. The increased risk of catastrophic failure of aging aircraft due to the detrimental effects of corrosion and its potential interactions with fatigue will be determined. The above analysis methods will be based upon both deterministic and probabilistic approaches.

6.3 Repairs

Structural repairs become more prevalent as aircraft age. This is especially true for aircraft which exceed their original design service lives. Special attention is being given to the adhesively-bonded composite repair of metallic structure. This type of repair was used to solve the C-141 weep-hole problem previously described and schematically shown in Fig. 6. Research is in progress to develop repair procedures and analysis methods for ensuring the integrity of composite repairs of metallic structures. In the past, repairs have primarily been based on static strength and stiffness requirements. Advanced technology is being developed that is based on more current durability and damage tolerance concepts (e.g., fracture mechanics). Analysis methods will be developed for ensuring the integrity of the damaged metal (i.e., subsequent fatigue crack growth), the adhesive bond (i.e., interface of composite patch and damaged metal), and the composite patch (i.e., detrimental effects of low-velocity impact damage). The effects of multiple repairs, environment (e.g., temperature), and disbonds on structural integrity are also being addressed.

6.4 Dynamics

Technologies are being developed which address dynamic issues in the areas of aeroelasticity and structural dynamics. Such technologies include advanced predictive methodologies and active suppression techniques. One problem receiving significant attention is buffet, which commonly occurs for twin-tail fighters (e.g., F-15) at high angles of attack. Vortex flow from leading-edge extensions can result in buffeting, causing severe dynamic loads on the twin vertical tails which can result in subsequent fatigue cracking problems. Such a buffet condition for an F-18 fighter is illustrated in Fig. 7. Another problem currently being investigated is limit cycle oscillations that are due to hanging stores under the wings of fighters (e.g., F-16). Dynamic issues dealing with unsteady/separated flow, shocks, vibrations and vortices are also being addressed. Acoustics and sonic fatigue are other dynamic areas which are receiving considerable attention. Advanced predictive methodologies and active suppression techniques are also being developed in these areas. Special attention is being paid to severe weapons bay acoustics, such as experienced by the B-1 bomber aircraft. Aeroacoustic loads and the detrimental effects they can have on aircraft structural integrity are also being investigated.

6.5 Health Monitoring

Structural health monitoring has high potential payoffs in terms of both safety and economics. On-board sensors are being developed for detecting the presence of structural damage and severe environments for aging aircraft. Such sensors are sought that can detect the presence of corrosion, corrosive environments, and fatigue cracks (e.g., multiple-site and/or multiple-element damage). Typical sensors are schematically illustrated in Fig. 8. Sensors are also sought that can ensure the integrity of structural repairs (e.g., monitor fatigue crack growth in damaged metal, the effectiveness of adhesive bonds, and the detrimental effects of low-velocity impact damage to composite patches). If such on-board sensors were available today, they would be useful for detecting corrosion in the C/KC-135 aircraft and ensuring the weep-hole repair integrity of the C-141 aircraft, issues previously discussed in this paper. The subject on-board sensors would be especially useful for structural locations that are inaccessible for inspections using conventional NDI techniques. The sensors could preclude expensive and time-consuming structural disassemblies currently required to perform such conventional inspections. The use of on-board sensors could result in significant savings in terms of cost, manpower and down time.

7. NONDESTRUCTIVE EVALUATION/INSPECTION

The Nondestructive Evaluation/Inspection technology category consists of two sub-categories: Corrosion Detection and Multi-Site Damage Detection (Fig. 9). Current methods (e.g., radioscopy) are being enhanced and transitioned to detect material losses that are less than 10% of the thickness due to corrosion. More novel, innovative methods are also being developed to detect nascent corrosion (i.e., corrosion at its very early stages of development). Corrosion detection in difficult-to-inspect areas (e.g., multiple-layer structures) is also being emphasized. This early corrosion detection will allow remediation prior to significant material loss, resulting in less expensive repairs. Likewise, current methods (e.g., scanning eddy current) are being enhanced and transitioned to allow rapid, large area assessments of the presence of fatigue cracks, with special emphasis on multi-site damage. More novel, innovative methods are being developed to detect very small cracks in difficult-to-inspect areas (e.g., multiple-layer structures). This will provide the capability to detect fatigue cracks at their very early stages of development, allowing remediation prior to significant crack link-up and, consequently, less expensive repairs.

8. AVIONICS

The Avionics technology category is subdivided into three technology sub-categories: System Avionics, Targeting and Attack, and Electron Devices (Fig. 10). A primary focus of the System Avionics research involves integrated modular avionics. An integrated communication/navigation/ identification system (ICNIS) is being developed for currently fielded weapon systems (e.g., F-15 and F-16). ICNIS is leveraging off the major investment that was made for the F-22. Special emphasis is being given to computational obsolescence (e.g., obsolete computer hardware and software). Affordable commercial off-the-shelf (COTS) technologies are being considered for solving aging system avionics problems. Rapid modification and testing of embedded avionics software is also being investigated under the System Avionics sub-category. The Targeting and Attack sub-category primarily involves the design, fabrication and laboratory testing of radar systems and aperture technologies. The main focus of the Electron Devices sub-category is on parts non-availability (i.e., replacement parts that are no longer available). Automated re-engineering research is being conducted to solve the parts non-availability problem. The Electron Devices sub-category also includes advanced solid state technologies that are being developed to replace obsolete technologies in the areas of electronics, microwave devices, microelectric sensors and microactuators.

9. PROPULSION

The Propulsion technology category consists of two technology sub-categories: Engines and Auxiliary Power (Fig. 11). For the Engines sub-category, technologies are being developed to reduce the risk of turbine engine failures. Such failures can be caused by variations in manufacturing quality, occurrences of foreign object damage (FOD), and changes in mission usage. A capability is being developed to assess the risk involved in reusing existing families of aging aircraft engine components. A capability is also being developed to perform real-time engine health monitoring to eliminate engine catastrophic failure. Materials compatibility tests are being conducted and filter coalescers are being developed that will allow the use of JP-8+100 fuel in aging aircraft engines. This will result in significant savings in fuel and maintenance costs. Additional research in the Engines sub-category will result in the design of thin dense chrome bearings that have longer lives and are more corrosion resistant than existing bearings. A primary focus of the Auxiliary Power sub-category is on maintenance-free batteries. No maintenance will be required for these batteries during their extended lives of up to 20 years. Significant maintenance and replacement cost savings will be realized over the lives of these batteries. Additionally, technologies are being developed to eliminate hydrazine from emergency power system components.

10. SUBSYSTEMS

The Subsystems technology category consists of four sub-categories: Mechanical Components, Transparencies, Flight Control and Cockpit Control/Displays (Fig. 12). The Mechanical Components sub-category includes technologies that result in the design and demonstration of improved mechanical components (e.g., landing gear structural components). Such components may be manufactured from advanced materials (e.g., titanium matrix composites) that are more durable and corrosion-resistant. Also, analytical and test methods are being developed for assuring the extended lives of tires. For the Transparencies sub-category, new technologies will provide more durable, hazard tolerant and cost effective transparencies that will result in significant cost savings over the lives of the transparencies. For the Flight Control sub-category, new technologies will allow the replacement of hydraulic actuators with electric actuators, eliminating dependence on the central hydraulic system. This will result in reduced flight-line maintenance requirements and aircraft down time as well as reduced depot time and repair costs. Also addressed in the Flight Control sub-category are advanced technologies that improve flight control maintenance diagnostics. This will enable a non-expert flight-line maintenance technician to fault isolate failures or discrepancies. This added capability will aid in wiring diagnostics and be applicable to all systems without aircraft modification. Finally, under the sub-category of Cockpit Control/Displays, advanced technologies will replace outdated and obsolete technologies to provide cockpit control/displays that are more modernized, standardized, reliable, safer and cost effective.

11. TECHNOLOGY ROADMAPS

Technology roadmaps have been developed for each of the technology sub-categories described in this paper. A total of 35 roadmaps currently exists. A typical roadmap for the Corrosion/Fatigue sub-category of the Structural Integrity technology category is shown in Fig. 13. Research efforts are shown for AFOSR's 6.1 basic research (e.g., Program Element 61101), Wright Laboratory's 6.2 exploratory development (e.g., Program Element 62201), Wright Laboratory's 6.3 advanced development (e.g., Program Element 63211), supplemental funding to Wright Laboratory from outside organizations (e.g., ASC) and related efforts of outside organizations (e.g., FAA and NASA). Wright Laboratory efforts include those that are within the projected budget (i.e., Wright Lab Funding) and those that are needed but for which funding has not been identified (i.e., Over-Ceiling Requirements).

12. TECHNOLOGY INVESTMENT

Wright Laboratory's total investment in 6.2 exploratory development and 6.3 advanced development for aging aircraft systems is presented in Table 1. Funding is identified for each of the five technology categories previously described, as well as the total for the five categories. A total 6.1 and 6.2 aging aircraft systems investment of \$51 million and \$223 million is being made for fiscal year (FY) 96 and FY96-00, respectively.

13. CONCLUSIONS

Technology development that supports aging aircraft is of the utmost importance to the United States Air Force. Many aircraft have already met or exceeded their original design service lives; many of these aircraft will considerably exceed their original life goals before they are removed from the inventory. Technologies are needed to ensure that these aging aircraft can be operated in a safe and economical manner. In order to ensure that these technologies are developed, the USAF formed the Aging Aircraft Systems Customer Focused Integrated Product Team (CFIPT) under Wright Laboratory leadership. This team formulated the Air Force Aging Aircraft Systems Program and identified the critical technologies that are described in this paper.

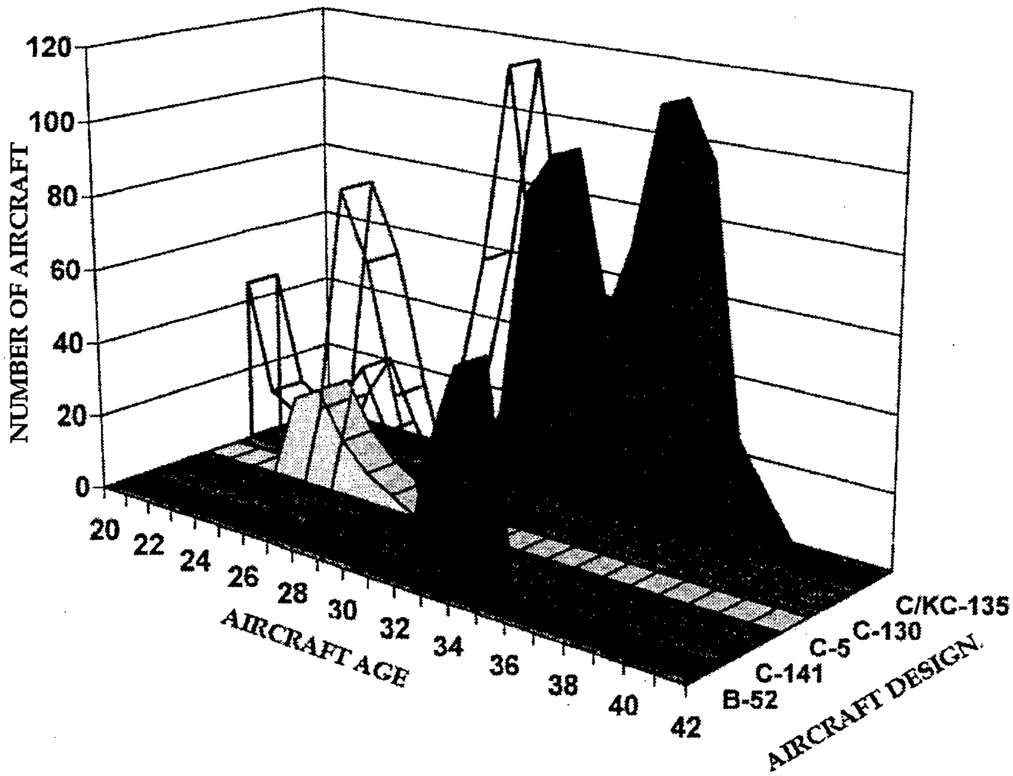


Figure 1. Inventory of USAF Larger Aircraft

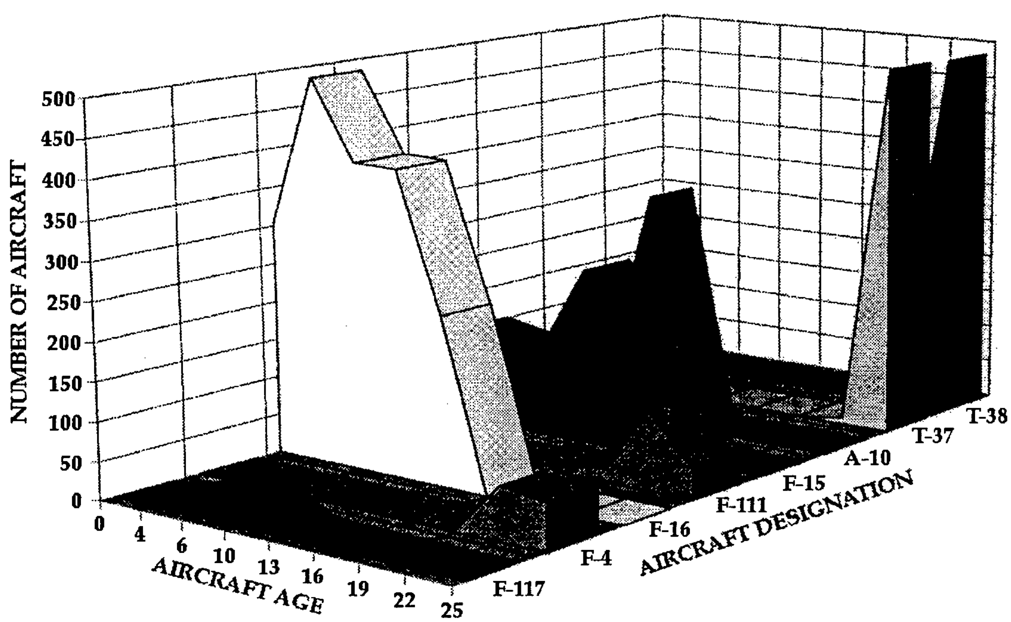
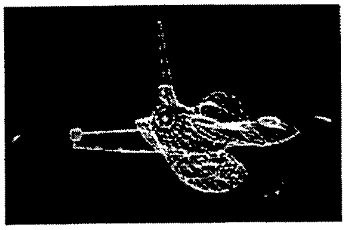
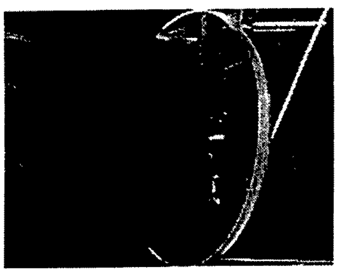


Figure 2. Inventory of USAF Smaller Aircraft



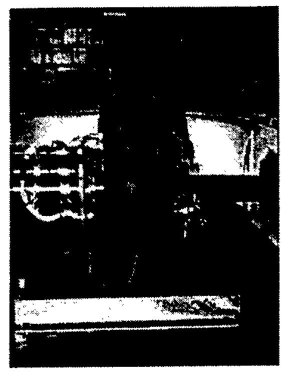
STRUCTURAL INTEGRITY

NDE/I



AVIONICS

PROPULSION



SUBSYSTEMS

Figure 3. USAF Aging Aircraft Systems Technology Categories

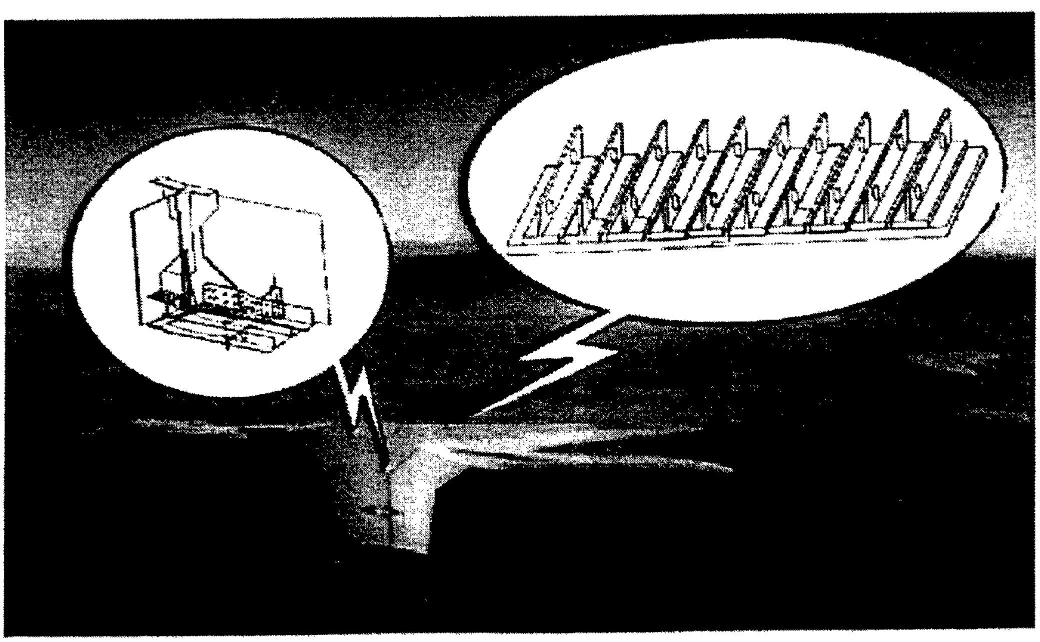


Figure 4. Widespread Fatigue Damage

Figure 5. Corrosion/Fatigue

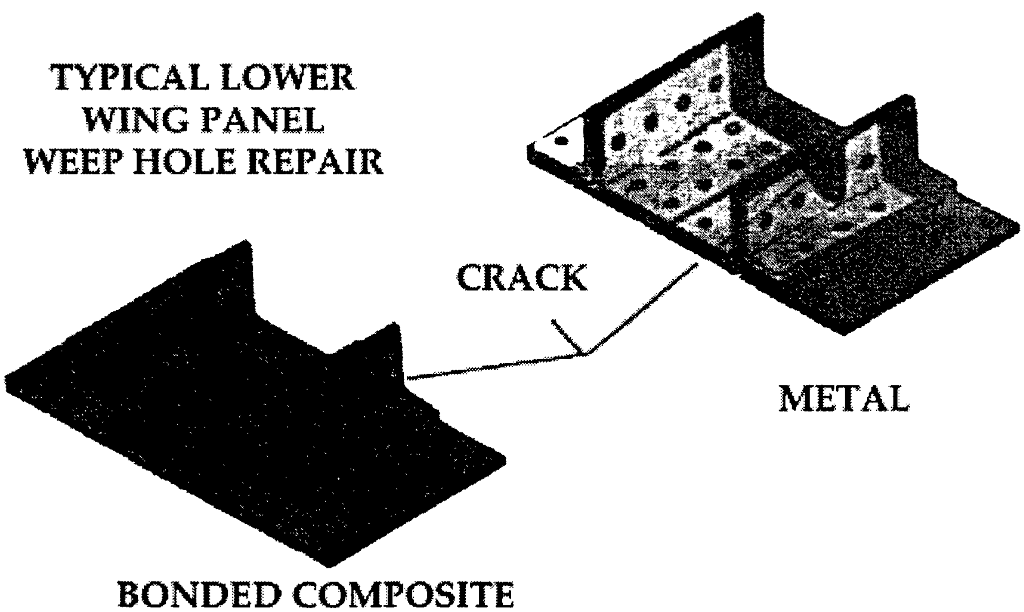


Figure 6. Repairs

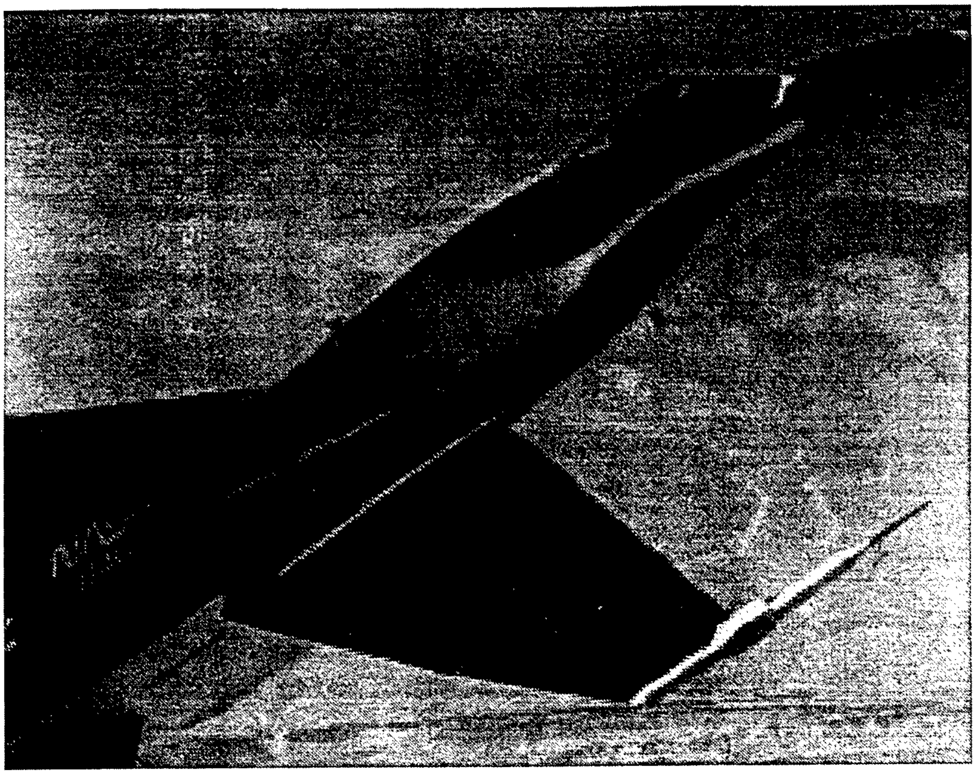


Figure 7. Dynamics

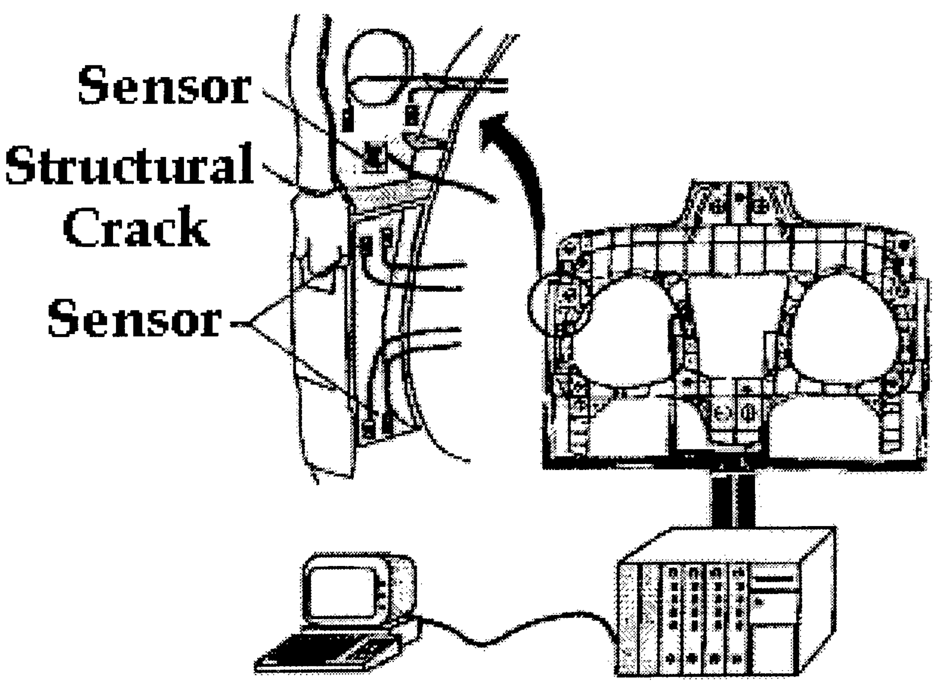
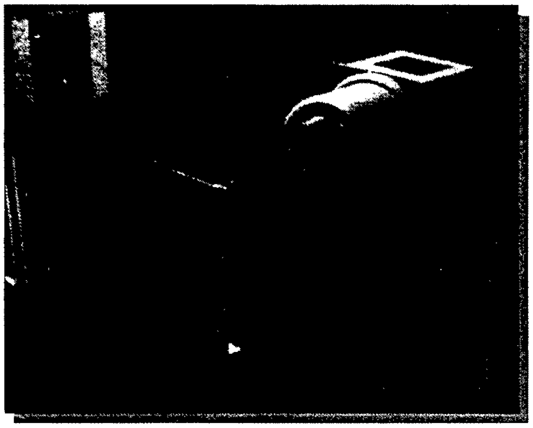
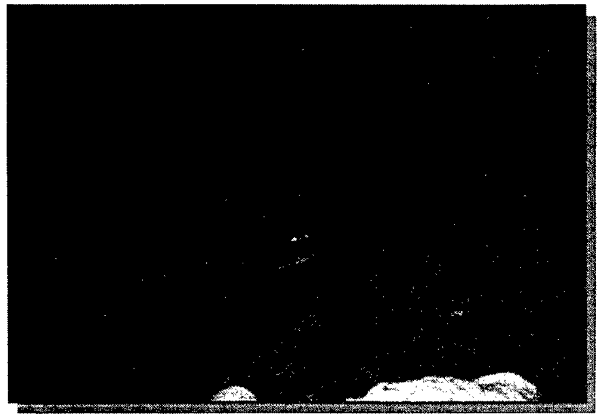


Figure 8. Health Monitoring



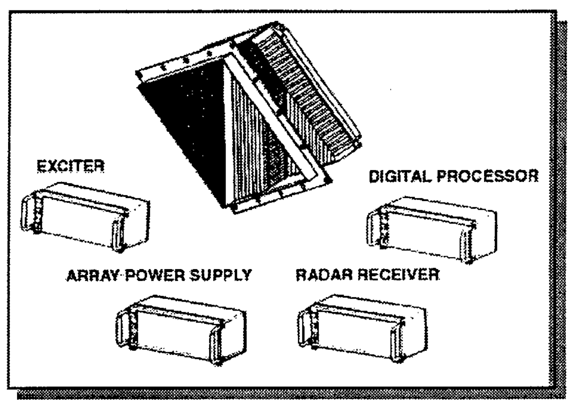
CORROSION DETECTION



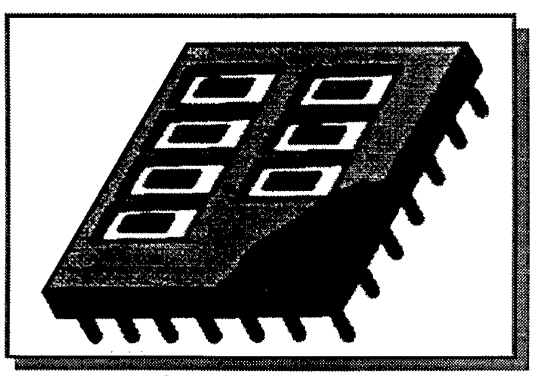
MULTI-SITE DAMAGE DETECTION

Figure 9. Nondestructive Evaluation/Inspection

SYSTEM AVIONICS

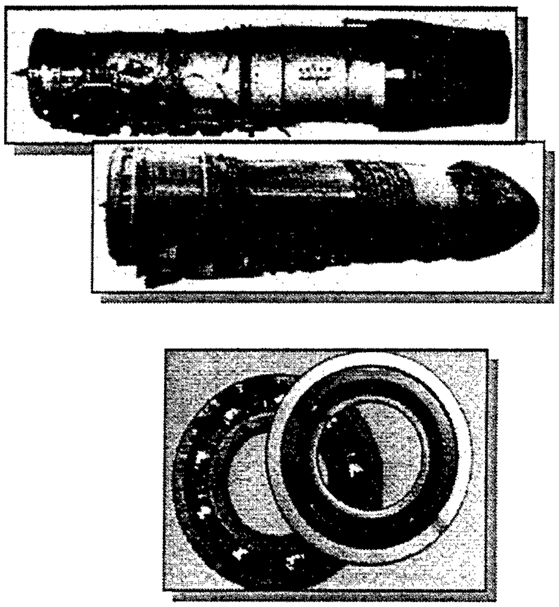


TARGETING AND ATTACK

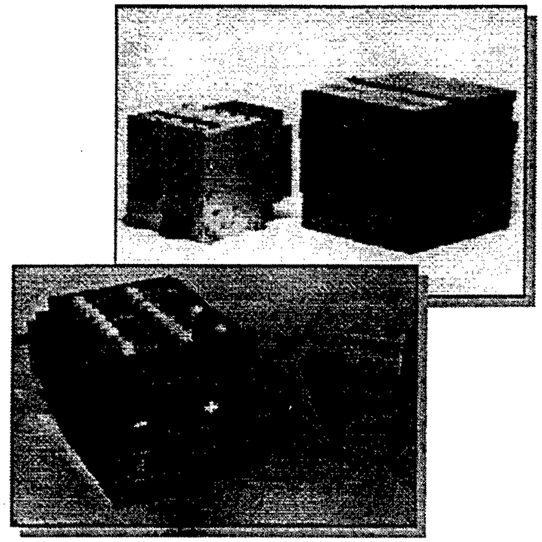


ELECTRON DEVICES

Figure 10. Avionics

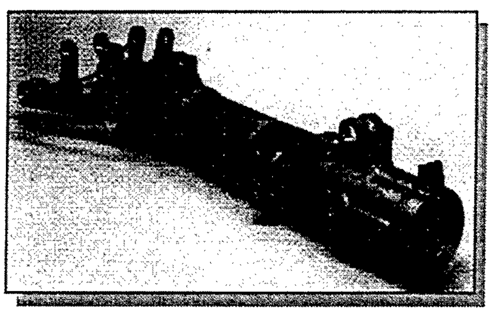


ENGINES



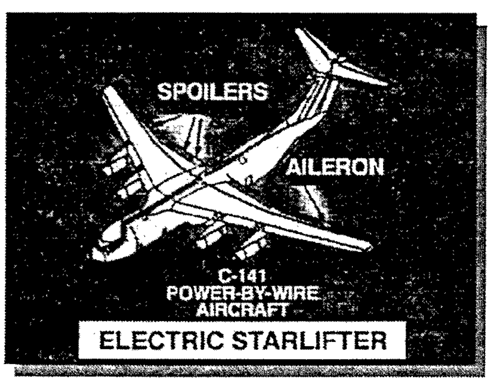
AUXILIARY POWER

Figure 11. Propulsion

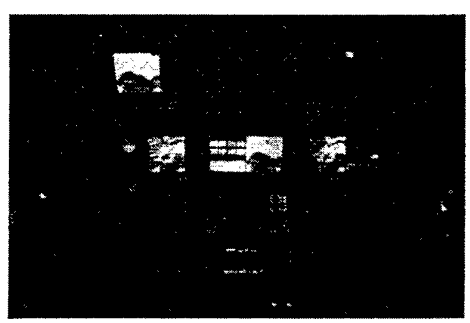


MECHANICAL COMPONENTS

TRANSPARENCIES



FLIGHT CONTROL



COCKPIT CONTROL/DISPLAYS

Figure 12. Subsystems

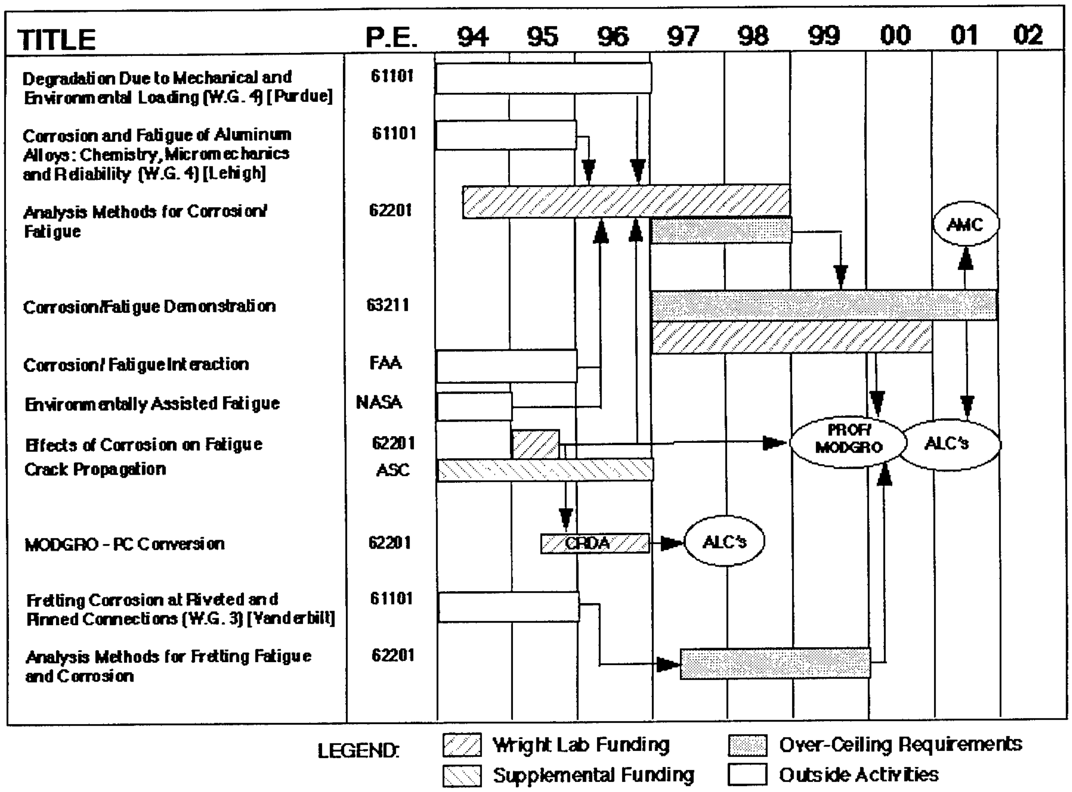


Figure 13. Corrosion/Fatigue Technology Roadmap

Table 1. 6.2 & 6.3 R&D Funding ($K)

TECHNOLOGY CATEGORY	FY96	FY96-00
Structural Integrity	12,499	73,522
NDE/I	4,008	24,604
Avionics	12,839	59,694
Propulsion	8,384	30,505
Subsystems	13,382	34,383
Total	51,112	222,708

AGEING AIRCRAFT - MANAGING THE TORNADO FLEET

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SUMMARY

1. During the 1980's the RAF procured a fleet of 228 Strike/Attack and 170 Air Defence Tornado aircraft to form the back-bone of its long term fleet of combat aircraft. At that time, the military threat against UK centred on a possible attack by the Warsaw Pact on mainland Britain. The Tornado aircraft was designed using 1970s technology, with a planned in-service life of 4000 flying hours and 100 Fatigue Index. The aircraft's original out of service date was 2003. Since then the primary use of the aircraft has changed to one of providing out of area support to United Nations and NATO operations. Furthermore, the aircraft's life has been extended so that it is now anticipated that the RAF will continue to fly the ADV until about 2010 and the IDS to approximately 2018, by which time individual in-service aircraft will have accrued some 9000+ flying hours. To meet this challenge the RAF has had to address how the airworthiness of the airframe and flight safety critical components can be assured, whilst at the same time, ensuring that the aircraft continues to provide an operationally effective weapon platform at a time of unprecedented, technological advances. This paper outlines the background to the Tornado project, details the various maintenance activities to ensure the aircraft's continued airworthiness for an extended life, and outlines the operational enhancements to ensure that Tornado remains a potent force capable of combatting modern threats in out of area theatres of operation.

INTRODUCTION

2. Good morning Ladies and Gentleman. Thank you for the opportunity to present the various measures that the Royal Air Force is adopting to manage its fleet of Tornado aircraft. From the outset, let me state that Tornado is the largest, operationally most important, and one of the newest fleets operated by the Royal Air Force; it was therefore something of a culture shock to be asked to give a presentation as to how the RAF is planning to keep its ageing fleet of Tornado aircraft in service for another 20 or so years. It rather reminded me of the day when, feeling youthful and energetic, I was playing cricket at my children's school in a match of dads against the schools 1st XI and, as I walked in to bat expecting to hit a century, I overheard the young bowler say we will soon get this old man out!

3. As I was introduced to you, I am Group Captain Martin Marlow-Spalding and my official title is Deputy Director of Support Management of Tornado. In summary, I am responsible for all aspects of supporting Tornado in-service and as such I head up a Multi-Disciplinary Group (MDG) to manage the fleet.

4. During my presentation I will briefly outline my responsibilities as the Tornado Multi-Disciplinary Group

Paper presented at the AGARD SMP Lecture Series on "Aging Combat Aircraft Fleets — Long Term Applications", held in Madrid, Spain from 7-8 October 1996; Pomezia, Italy from 10-11 October 1996; Atlanta, USA from 4-5 November 1996 and Brussels, Belgium from 22-24 January 1997, and published in LS-206.

Commander so as to give you a feel for the way that the RAF has brought together and then delegated the wide ranging authority. I will then outline the scale and size of the Tornado Tri-national project where I will detail the planning and design aspects, the production phase and the different roles that Tornado undertakes around the world. I will then cover the RAF fleet of aircraft in some detail before moving on to look in detail at the activities that are planned to keep the aircraft in-Service until its Out-of-Service date. In concentrating on this latter phase, I will discuss separately our management of the IDS ground-attack aircraft and the Air Defence Variant. I will look at the philosophy of fatigue management, fatigue testing, and our current and future aircraft structural programmes; I will then outline the work we have set in hand to requalify components and equipments fitted to the aircraft before reviewing the operational enhancements that we have planned for both fleets.

5. My presentation will last approximately one hour leaving a period when I look forward to answering any questions that you may have. However, if anybody wishes to seek clarification during my presentation, please do not hesitate to interrupt me.

ROLE OF MULTI-DISCIPLINARY GROUPS

6. Prior to 1990 the management of the Royal Air Force was structured on a functional basis with Aircrew answering to the Chief of Air Staff, Engineers working to the Chief Engineer(RAF), Suppliers responsible to Director General of Supply Management, Financiers working for the Treasury, and Contract staff answering to their own Director of Contracts. There was no co-ordinated in-service Project Management based on aircraft types; we worked from separate locations, often hundreds of miles apart and it took an eternity to take our business forward. The only dubious advantage was that as an engineer I could always blame the suppliers, financiers or contracts staff if there was a lack of spares to fix my aeroplanes! However, in 1990, the whole organisation of the RAF changed with the launch of the Multi-Disciplinary Group (MDG). These Groups are based on specific aircraft types, in my case Tornado. The Group has a single purpose and consists of a Team encompassing the aircrew, engineers, suppliers, financiers and contract staff co-located and all working for me in direct support of Tornado. In total, I have a staff of 150 personnel consisting 90 suppliers, 40 engineers, 5 financiers, 2 airmen, and 12 contract staff made up of a mix of service and civilian personnel. I do not have responsibility for the engine and one or two common items of equipment which are handled by specialist commodity MDGs collocated at RAF Wyton. I control an annual budget of approximately £200M ($300M), and I have considerable financial authority; I can commit up to £5M ($7.5M) for any single contract. I am responsible for the overall airworthiness of the fleet, and am required to provide sufficient aircraft available in the right configuration at the front line to meet both operational and routine training commitments; finally, I have total responsibility for the timely procurement of the 285,000 lines of spares used on the aircraft. The day-to-day management of the aircraft on the Main Operating Bases and Deployment Bases is managed by Headquarters Strike Command, but I am responsible for programming aircraft into the third and fourth line levels of maintenance at the RAF's maintenance base at Royal Air Force St Athan and Industry

respectively. Gentleman, I have covered this topic in a little detail so that you can understand my responsibilities when it comes to looking at how we manage our in-service aircraft.

THE TORNADO PROGRAMME

7. Background. In 1969 it was agreed that the UK should have an all-weather, ultra low level capability for conventional, strike and reconnaissance operations by the early 1980s. It was recognised soon afterwards that certain maritime strike/reconnaissance and air-defence improvements would need to be introduced and that considerable savings in R & D and unit production costs would result if all these requirements were satisfied by suitably equipped versions of one basic aircraft. High standards and advances in performance, weapon delivery capability and serviceability were demanded. The result was a relatively small swing-wing aircraft powered by twin, new-technology engines and featuring advanced avionics and maintenance features; it was named the Multi-Role Combat Aircraft (MRCA).

8. In furtherance of the policy of co-operation within the European defence business, the UK, Germany and Italy agreed to develop the MRCA to a set of compromise performance parameters satisfying each Nation's main requirements and providing the basis for an air defence derivative. In 1970 the tri-national requirements were detailed in the Performance and Design Requirements (PDR); this specified a two-man crew, twin-engined aircraft optimised for use in the NATO area, initially in the low level strike and reconnaissance roles, but with potential for fighter development.

9. The three nations collaborated formally by controlling the programme through the NATO MRCA Development and Production Management Organisation (NAMMO) at senior executive level, whilst their policies were put into effect by the Management Agency (NAMMA), located in Munich and staffed by service and civilian government specialists from all three nations. NAMMA included elements of the air staffs, engineers, suppliers, finance and the contracts staff of the Procurement Executive and still runs certain aspects of the programme on a day-to-day basis on behalf of the sponsoring governments.

10. On the commercial side, an international company - Panavia - was formed, jointly owned by the three principal airframe concerns, BAe, Aeritalia (now Alenia) and Messerchmidt Bolkow Blohm (now Daimler Benz Aerospace). The company has its headquarters in Munich and employed about 400 direct employees; it served as a centralised management body, placing contracts on the three national companies who in turn sub-contract as necessary. Separate contractors dealing direct with NAMMA are Turbo Union (Rolls Royce, Fiat and Motoren Turbo Union) for the engine, and Mauser for the gun. Altogether some 70,000 people were involved in the manufacture of the Tornado.

11. Development and production was shared between the Nations roughly in proportion of the intended buy (385 for the RAF, 322 for Germany, and 100 for Italy). Broadly, the UK has been responsible for the front and rear fuselage, Germany for the centre fuselage and Italy for the wings. Engine modules were similarly shared between the partners. Complete sub-assemblies were delivered to separate production

lines in each of the 3 countries. Similarly, different parts of the development and clearance programme were allocated to each national contractor and Operational Test Centre (OTC), and the whole programme brought together through NAMMA as Release to Service recommendations. A tri-national exchange of flight safety information was also exercised by the national safety organisations in conjunction with NAMMA. This organisation lasted largely unchanged from initial concept, through design, development and production to in-service support. It is fair to say, however, that, as time has moved on, there has been a general shift for in-service issues to be handled nationally. Moreover, in 1995 the Agencies for Tornado and EF2000 combined to form the single NATO European EF2000 and Tornado Agency - NETMA. I have deliberately explained the background to the Tornado Project as it has played a significant part in how we have maintained and enhanced the aircraft in service.

12. <u>Production</u>. In all some 985 Tornado aircraft will have been built, including development and production aircraft, with the last one currently on order and due for delivery in 1997. Both the IDS and ADV variants were designed to a Mil Spec safe life philosophy with a life of 4000 PDR hours. The safe life had to be demonstrated by a fatigue test specimen. The aircraft have been delivered to 9 distinct batch build standards. The first production aircraft was completed in 1980 and at the height of production aircraft were being delivered to the 3 Nations at the rate of 10 per month. Both the ADV and GR1 have been built to two standards: single and twin stick trainer variants. The Royal Air Force has received 228 IDS (178 single stick/Strike and 50 twin stick/trainer aircraft), 170 ADVs (119 single stick and 51 twin stick/trainer aircraft), the GAF 357 IDS (290 single stick and 67 twin stick/trainer aircraft), the IAF 99 (87 single stick and 12 twin stick/trainer aircraft), the RSAF 96 IDS (72 single stick and 24 twin stick/trainer aircraft) and 24 ADVs consisting of 18 single stick and 6 trainer F3s. In addition, the RAF has in the past year leased 12 ADVs to the Italian Air Force with a further 12 due to be leased starting in January 1997.

13. <u>Types/Roles</u>. Having stated that the production run totalled nearly 1000 aircraft, let me now turn to the various Air Forces operating the Tornado and the primary variants of the aircraft which are almost as varied as the venerable Phantom F4. Fundamentally there are 2 variants of Tornado; the Strike/Attack IDS and the F3 fighter aircraft. The ADV is some 1390 mm longer than the IDS and contains a number of structural differences in the wings and fuselage and has a significantly different avionics fit. The German Air Force operate the IDS whilst the Royal Air Force, the Royal Saudi Air Force and the Italian Air Force also operate the F3. The IDS aircraft come in a number of sub-variants including the RAF operated GR1A and GR1B reconaissance and maritime attack aircraft; the reconnaissance aircraft is fitted with a left and right hand side-loading infrared system and an IR Linescan whilst the GR1B has been optimised to carry the Sea Eagle anti-shipping missile. In addition, the Royal Air Force has modified a number of its GR1s to carry the Air Launched Anti-Radiation Missile (ALARM) which, although employed in a similar role as the High-Speed Anti-Radiation Missile (HARM), is a fundamentally different missile. In addition, a number of our IDS aircraft

can carry the GEC Thermal Imaging Airborne Laser Designating (TIALD) Pod which compares with the Lantirn pod, whilst others are equipped with Night Vision Goggles, an enhancement procured from Oxley Developments. The F3 fleet is simpler with one basic variant capable of carrying 2 external 2250 litre fuel ranks, 4 Skyflash MRAAMs, 4 Sidewinder AIM 9L SRAAMS, and a single 27mm Mauser Canon, although again there are currently a number of mini-fleets to cover the Joint Tactical Information Distribution System (JTIDS), Towed Radar Decoy, Night Vision Goggles and Chaff and Flare. Within the next 2 years we plan to bring all of our F3s up to a common, operational standard. In the longer term, we plan to modify both the IDS and F3 to bring them up to an enhanced operating standard to take them into the 21st Century. I will discuss these programmes in some detail later as part of our strategy to manage an ageing fleet. Turning to the other Nations, the Germans have introduced an ECR variant, which is equipped with an Emitter Locator System, the HARM anti-radiation missile and a reconnaissance suite of conventional and electro-optic sensors. The Italians are also modifying a number of their aircraft to fill an ECR role and also operate the Cormorant anti-shipping missile.

14. Basing. Finally, I thought that it might be useful if I briefly summarised our current basing strategy. For the GR1, basic training is carried out at the Tri-National Tornado Training Establishment at RAF Cottesmore where crews from Germany, Italy and the Royal Air Force learn to fly the Tornado. The RAF crews then transfer to RAF Lossiemouth where they undergo their operational and weapons training on XV(R) Sqn. From there, crews transfer to one of our eight operational squadrons; we have two GR1b maritime attack squadrons: Nos 12 and 617 Sqns, also based at RAF Lossiemouth. We have 2 GR1a reconnaissance squadrons: Nos 2 and 13AC Sqns based at RAF Marham, and 4 attack squadrons: Nos IX, 14, 17 and 31 Sqns based at RAF Bruggen in Germany where they operate in the ALARM, TIALD and Vicon recce role. In addition, since the Gulf War, we have a Flight of GR1s operating from Incirlik in Turkey and Dhahran in Saudi Arabia as part of the multi-national force enforcing the UN Northern and Southern No-fly zones of Iraq. Turning to the F3, crews learn to operate the aircraft at the Operational Conversion Unit at RAF Coningsby before transferring to one of 6 operational squadrons; we have 2 squadrons based at each of the 3 ADV Main Operating Bases at RAF Coningsby, Leeming and Leuchars. We also have a Flight of F3s based in the Falkland Islands and, until recently, a Flight based at Gioia Del Colle in Italy supporting the UN No-fly zone over Bosnia - Hertsogovena.

STRUCTURAL INTEGRITY

15. Relationship Between Test Specimen and In-Service Lives. Before looking at how the RAF intends to manage the long term fatigue life of the aircraft let me look at the relationship between Airframe Fatigue Specimen and In-Service Lives. Tornado was influenced by the US Mil Spec which required a scatter factor of 4 for the fatigue test. Thus 1600 TH were required in order to demonstrate safe life of 4000 FH. However, the UK CA release (DMCA release) requires different scatter factors; we apply a safety factor of 3.3 for structure that is monitored by a Fatigue Meter Formula (FMF) and a factor of 5 for structure that

is not monitored by the various fatigue formulae in order to demonstrate a safe life of 4000 FH for UK aircraft, the test specimens must clear:

a. 100FI for monitored structure which equates to 13300TH for a 4000 in-service airframe,

b. 20000TH for unmonitored structure lifed in FH,

c. And 16000TH for monitored structure lifed in FI.

The above figures are based on the principle that data from our structural usage monitoring aircraft have not demonstrated a significantly different damage rate to that predicted by the design authority.

16. Monitored structure is defined, that structure for which a clear relationship between vertical acceleration (Nz) and loading has been established, and for which the rate of damage can be approximated using an FMF. Unmonitored structure is that structure for which no clear relationship between vertical acceleration (Nz) and loading has been established, or for which the rate of damage cannot be approximated using an FMF. Fatigue damages found on the test specimens are analysed to find the particular associated loading actions. On less complex aircraft, damages can often be attributed to similar loading actions; however, as a result of modular construction and variable geometry design, Tornado damages can be associated with one of several loading actions. So to enable accurate monitoring, Fatigue Monitoring Formulae (FMF) are derived to calculate the life consumed of the unmonitored structure. Tornado use 7 Fatigue Monitoring Formulae; the Frame 8000, Frame X9110, the Pivot Fuselage, the Duct, the Pivot Wing, the Wing Carry Through Box (WCTB), the WCTB Aft Link (ADV only) and the Forward Fuselage Vertical Bending (ADV only). The FMF can be broadly related by ratio, such that based on the lead formulae, that of Frame 9110, the Pivot Fuselage factor is 0.67, the Wing Pylon factor is 0.73, the Duct factor is 0.38, and the WCTB factor is 0.77. However, I should stress that they depend in part on the time spent at each wing-sweep angle (Stress per Bending Moment Ratio). The main point that I wish to make is that the ratios vary from aircraft to aircraft and fatigue consumption has to be calculated for each aircraft and for each sortie.

17. <u>Fatigue Index</u>. The life of an aircraft structure is generally quoted as a set number of flying hours (FH) at the rate of usage specified in the original design lead spectrum. Hence, for the Tornado IDS, the aircraft was designed for a safe life of 4000 FH within the PDR loading spectrum. Thus it is assumed that after 4000 FH of design usage, the fatigue life of the airframe will have been consumed. If, however, the airframe is operated outside the design spectrum, the rate at which it's fatigue life is consumed will differ from the rate at which FH are consumed. Thus, it is necessary, in terms of both safety and cost-effectiveness, to measure the consumption of fatigue life to ensure that the safe life is not exceeded, and that aircraft are not retired prematurely. In order to gain a measure of the rate at which this fatigue life is consumed, each aircraft is given a Fatigue Index (FI) of 100 when it is delivered. The consumption of this FI is calculated by measurement of Nz, aircraft mid-sortie mass, stores configuration, sortie duration and sortie pattern, and for an

aircraft that flew strictly to the PDR spectrum throughout its life, 100 FI would be consumed in 4000 FH. However, the rate of actual FI consumption varies from aircraft to aircraft and sortie to sortie, and it is probable that many aircraft will reach 100 FI before 4000 FH.

18. Let me now look at the various main fatigue tests starting with the full-scale Hinge Fatigue Test.

19. Full-Scale Hinge Fatigue Test (FSH FT). This test was used to prove the life of the prototype wing and variable geometry design. A production wing was tested until failure which occurred at 2600TH (16.25FI Wing Pylon). The failure was confirmed on the Main Airframe Fatigue Test when the RH wing failed in the same place at similar test hours. As a result modifications were introduced to cold work the wing lower spar boom and lower skin. Moreover, build changes were introduced that replaced the titanium round rib bolts with items manufactured from steel from the 44th wing set onwards.

20. Major Airframe Fatigue Test (MAFT). The primary fatigue test specimen is the MAFT, which is a tri-nationally funded project based at IABG Ottobrunn, run by DASA, to qualify the IDS structure The test started in 1981 and reached 16000TH in 1990. It is currently at 18000TH and is expected to reach 20,000TH by the end of 1996. The majority of the structure is qualified to 100FI, but some areas including the undercarriage back-up structure, areas of front and rear fuselage, and some modification and repairs, require additional test evidence. Moreover, many areas of structure are now strain-gauged to allow read-across from MAFT to in-service loading. Early failures to the intake duct occurred at 2600TH with failures to the F8000 intake ring occurring shortly after. This resulted in the first retrospective fatigue programme, the 5FI programme, which was applicable to the first 199 aircraft from the production line. The 5FI programme totalled about 3000 manhours per aircraft. Further minor fatigue programmes were required on early batch aircraft at F9110 at 10FI and at 16FI on the wing formulae to cold work the lower wing plank and to improve the strength of the pylon cut-out and rear spar. Following these early, unwelcome structural programmes, subsequent testing of MAFT proved that the aircraft had safe life of approximately 5000 flying hours or 50 FI on the F9110 formula without any further intervention. However, to extend the aircraft to its out of service date we will need to carry out a major modification programme akin to the ADV mid-life fatigue programme that I will discuss later. Furthermore, we will need to extend MAFT testing to 45000 test hours if we are to clear the unmonitored structure using a X5 safety factor; this requirement is unlikely to be achieved and we will need to devise alternative clearance procedures. Work has started on this initiative but our analysis is still at an early stage of development.

21. ADV Main Airframe Fatigue Test (ADMAFT). The ADV Main Airframe Fatigue Test (ADMAFT) is based at BAe Warton and is run by BAe, to qualify the ADV structure; testing started in mid-1986 and is currently at 11950TH. The test is expected to reach 16,000TH by mid 2000. Progress to date has been delayed by major structural damages found at 4000TH and further damage at 10,500TH. In each case, critical damage occurred in the intake duct panels and in the area of frames X10545 and X10910. The damage at

4000TH resulted in the definition of the 25FI (Mod 14059) fatigue programme. The programme consisted principally of cold-working and fitting oversize fasteners to a number of fuselage frames as well as slot-peening certain areas and adding reinforcing elements to other parts of the structure that had been found to be critical. The work content of the 25FI programme, including strip and recovery of the aircraft is about 11000 manhours with the core modification taking about 6000 hours. The failure at 10500TH will necessitate a greater level of intervention with numerous centre fuselage frames and skin being cold-worked, replaced or strengthened. The work will form the core of the mid-life fatigue programme which is due to start in about 1999. Embodiment will probably require the centre fuselage to be split out, not a particularly difficult task, and replaced with either a build jig or a new, substantial support jig.

22. ADV Wing Clearance. ADMAFT has also been used to clear the ADV wing structure. A failure of the Front Spar Closure Plate (FSCP) and lower diffusion area web at approximately 65FI on the Wing Pivot formula, which equates to approximately 98FI on the X9110 formula. However, we may be able to extend the wing to about 72FI (Wing Pivot) by cold-working the web and changing the Front Spar Closure Plate material. Furthermore, the Wing Carry Through Box (WCTB) failed on ADMAFT when its shear wall had a catastrophic failure at 47FI on the WCTB fatigue formula. The WCTB is now planned to be replaced as part of the ADV Mid-Life Fatigue Programme.

23. ADV Wing Accelerated Fatigue Test (AWAFT). In addition to ADMAFT, the AWAFT was used to demonstrate the clearance of ADV wings which had benefitted from in-line solutions to the Full Scale Hinge Fatigue Test, the Retro-wing Fatigue Test and MAFT wing damages. AWAFT was also used to qualify a package of Modifications planned to be embodied at 32FI wing as a result of ADMAFT wing damages to the in-board Pylon Cut-Out area and the lower wing skin. The aim was to confirm that the wing could be extended from 32FI towards 72FI. However, this fatigue test failed in Apr 95 when the wing fractured under the proposed 32FI Butt-Strap, thereby demonstrating that there was no fatigue benefit over pre-Mod wings. Therefore, AWAFT has demonstrated a life of 72FI in the wing in-board pylon areas, subject to Mod 25011 being carried out at approximately 38FI wing formula and confirmation by a second AWAFT together with the modifications to the front closure plate and cold working the lower diffusion web.

24. Fatigue Management. In order to manage FI consumption, and to ensure that the cleared flight envelope is not inadvertently exceeded, FI consumption is monitored for each aircraft sortie using data collected from a counting accelerometer which records the number of exceedances of preset vertical acceleration (Nz) values in three wing sweep bands: 24, 45 and 67 Deg. Sortie duration, sortie profile and external stores configurations are recorded and the mid-sortie mass calculated. These figures allow accurate estimation of FI consumed, per aircraft, per sortie. These are recorded for every airframe, wing, taileron and WCTB and analysed and published monthly. The Tornado Fleet Manager and the Air Staff use this data to set "Fatigue Budgets" for each unit, based on a balance between operational and training requirements and aircraft availability. When necessary, individual

aircraft can be "managed" into fatigue enhancement programmes by identifying aircraft to slow-fly or fast-fly, thereby ensuring that fatigue consumption is optimised. However, care is needed to ensure that aircraft are phased into programmes early enough to avoid grounding aircraft due to be floor-loaded later in the programme, but not so early as to "waste" FI on those aircraft at the front of the programme.

25. Structural Usage Monitoring System (SUMS). The fleet fatigue monitoring system is calculated on loading estimations based on FEM and design assumptions. SUMS data is then used in order to refine the data gathered. SUMS is fitted to 8 in-service Tornado (5 IDS and 3 ADV) and records in-flight data from a range of transducers and in-service sensors. The data captured includes: Airspeed, Altitude, Acceleration in all three axes (Nx, Ny, Nz); Load-calibrated strain gauge bridges; uncalibrated "hot-spot" strain gauges and digital information on control surface positions etc. from the Data Acquisition Unit fitted to the aircraft accident data recorder. The data is analysed by BAe for every SUMS sortie and generates correction factors to refine the FI consumption calculations.

26. Other Structural Damage. In addition to the main fatigue programmes, we also rely on directed inspection of specific areas and repair on defect. These areas include:

a. Cracking in the Main fitting and soft angles, the pintle frame of the main landing gear, as well as the nose landing gear.

b. Cracking of the land of the wing tank access panel.

c. Cracking from adjacent fasteners in the intake duct skin.

d. Corrosion in a few specific areas, including the lower wing plank, fuel tank floors, and a patch at the base of the fin.

e. Accidental damage caused by bird-strikes, slip of the hand, ground collision.

f. Environmental damage, ie corrosion.

g. Fatigue damage outside the specimen experiences as a result of loads not simulated on test ie overstress.

In order to ensure the structural integrity (SI) of the aircraft, regular scheduled maintenance is carried out, including directed inspection of Structurally Significant Items (those items whose failure could cause the loss of an aircraft or its crew, or would result in a significant reduction in structural strength), and zonal inspections of the remainder of the aircraft.

27. Structural and Conversion Sampling. As it is not possible to inspect all structural significant items regularly without dismantling parts of the aircraft structure, the scheduled maintenance programme is augmented by periodic structural sampling. This sampling programme is akin to the US ACI "Queen of the Fleet" programme, where fleet leader aircraft are periodically examined for fatigue, corrosion and accidental damage in areas not normally inspected. Moreover, in order to minimise the threat to structural integrity posed by corrosion, those aircraft which operate in a maritime environment are subjected to regular corrosion surveys. The

results from these surveys, as well as allowing timely treatment of corrosion, go to build up a corrosion database. This allows trends to be monitored and preventative programmes to be formulated.

28. Tornado Structural Integrity Database (TSID). In the wake of the Aloha Boeing 737 incident, MASAAG 83 recommendations included a requirement to carry out Ageing Aircraft Audits and to monitor repairs, corrosion blends and Multi-site Damage issues. In order to comply with that requirement, TSID is being developed to collate the information obtained from structural sampling and corrosion surveys, and to record graphically the application of repairs and corrosion blends to every aircraft in the RAF fleet. Having concentrated on the airframe, let me now look at how we intend to extend the components.

COMPONENT LIFE EXTENSION

29. In turning to Life Extension of the components let me start with the engine. All RB199 engine Group A Parts have specific lives established which are continuously reviewed by the Engine Lifing and Qualification Committee. The lead RAF RB199 engine module has already reached the current qualification limitation of 3,000 engine flying hours and is now quarantined. The engine manufacturer, Turbo Union, has confirmed that all Non-Group A Parts, with the exception of 6 items, may remain in service and be maintained by application of the on-condition concept up to 7500 engine flying hours; these items do not require specific life extension programmes. The 6 items that require a life extension study are: the Combustion Chamber Outer Casing, the inner and outer intermediate casing, the IP stubshaft bolted joint, the LP Turbine Starter and the IP/LP Bearing Housing. An analytical study for life extension is currently being conducted under Post Design Task to provide a minimum life extension of 500 hrs pending further extension test programmes.

AIRCRAFT EQUIPMENT

30. This is being addressed in 4 phases:

a. Phase A - Identification of Safety Relevant Components.

b. Phase B - Assessment of the data proformae raised for Phase A components.

c. Phase C - Requalification activities, where required, for safety relevant items.

d. Phase D - Certification process for equipment and the aircraft.

This process superseded an initial process which would have involved requalification of all equipment whether or not it had any impact on flight safety or airworthiness; a process that would have been prohibitively expensive and have taken a minimum of 3 years to complete. The current approach involves a partnering relationship with the 3 Nations working in conjunction with Industry. Within the UK, the Tornado Support Authority and certain aspects of the RAF's Logistics Support Services are currently involved in an intensive series of meetings for the 18 systems which contain safety relevant items as identified by the Panavia Phase A report issued on 28 Mar 96. Each of these meetings is held at the respective SDR companies, and the list of safety-relevant items is being progressively reduced through a risk assessment review for each item. Germany and Italy are also involved in this assessment activity which should be completed shortly. Items identified as safety critical will either be replaced with new components or will be requalified to provide a new, extended, in-service life. Having started with something in excess of 800 equipments identified as being safety critical, the list has been reduced to 180 items, and each of these is currently undergoing the risk assessment process. However, all is not straight-forward, we have yet to devise a process where a lifed item, that cannot be requalified and does not have a log card, can be tracked and replaced at the end of its life. This issue is currently taxing our minds and I would welcome ideas from the floor if anyone has been faced with and resolved this issue.

31. Life Extension Summary. Although there is considerable work to be done in all 3 areas, namely structures, engines and equipments, we are on track to achieve aircraft life extension in time to meet our requirements of beginning this process in about 9 months' time. Much of the success so far is attributable to Industry's refreshing new willingness to satisfy the customer. This attitude is a marked improvement over our earlier attempts where Post Design Tasks were initiated but then had to be aborted in 1993 by the 3 Nations because of a lack of imagination from Industry to meet the customers' needs at a price that was affordable.

OPERATIONAL ENHANCEMENTS

32. IDS Tornado MLU - Background. In 1987 the RAF recognised the need to update some of the 1970 technology employed on its aircraft to maintain the Tornado operational capability until its planned out-of-service date which was then 2005. As a result, the MOD raised a task on BAe to outline their proposals. At that stage the Tornado GR1s primary role was part of NATO Strike-Attack force against the Warsaw pact, and much of the early design was built on the aircraft's requirement to attack, day or night, in all weather and at ultra-low level but within a pre-determined geographical area. Therefore, the initial design centred around making the aircraft capable of operating covertly and being able to combat enemy defences successfully. At the heart of the enhancement was the GEC Terrain Reference Navigation (TRN) system that could be used to augment the existing very capable but, in radar signature terms, noisy Terrain Following Radar. As a result of the break-up of the Warsaw Pact, combined with experience gained from Desert Storm, the Royal Air Force reviewed the technical content of the up-date programme and in 1993 issued a new specification known as MLU-93. The new specification was aimed at enhancing the aircraft's suitability for world-wide, out-of-area operations whilst retaining much of the original work with the exception of the Terrain Reference Navigation System.

33. MLU-93 Content. MLU-93 introduces 6 new major capabilities at the same time as combining 4 new staff requirements; these 10 elements include:

a. Mil-Std 1760 New Armament Control System. The existing Tornado GR1/1A Stores Management System (SMS), part of the Armament Control System architecture, is to be improved by the modification of the Weapons

Programming Unit (WPU) and Weapons Control Unit (WCP1). A Weapons Interface Unit, which comprises two LRUs, is also to be introduced. Wiring for a full Mil-Std 1760A Class 2 Weapon databus is provided to the eleven Weapon stations. The Weapon station pylons are to be modified accommodating both electrical and structural changes. The Missile Control Unit (MCU), the Air-to-Air Missile Unit (AAMU), one Gun Electronic Unit (GEU) and all other pre-modification standard Armament Control Unit LRUs are retained unchanged.

b. <u>Mil-Std 1553B Databus</u>. To facilitate the integration of new avionic systems, a 1553B Databus is introduced.

c. <u>Head-Up/Head-Down Display/FLIR</u>. The major new displays are the Pilot's Head-Up Display Unit (PHUD) and the Pilot's Multi-Function Display (PMFD). These displays together with the TV Tabs are driven by the Computer Symbol Generator (CSG). The CSG is a new unit introduced at MLU to replace the existing GR1/1A standard Waveform Generator (WFG) and HUD Electronics Unit (HUD EU). The map display is produced by the Digital Map Generator (DMG). The PFMD and DMG provide the Head-Down FLIR information.

d. <u>Enhanced Data Preparation/Loading</u>. A Computer Loading System which consists of a Transportable Data Module (TDM) and a Data Entry Unit (DEU) is installed to enable data loading of the freely programmable processors. The data is down-loaded from the TDM via the DEU and Mil-Std 1553 databus. The RAF is in the process of procuring a Tornado Advanced Mission Planning Aid (TAMPA) to meet the GR4 ISD of Sep 98, thus ensuring functionality between aircraft and planning aid software.

e. <u>Video Recording System</u>. This new system replaces the existing wet film HUD and displays recording systems. It comprises a Video Electronics Unit (VEU), a Video Tape Recorder (VTR) and a control panel. To enable ground replay of the recorded in-flight data, a Ground Replay Facility (GRF) is available which de-multiplexes the recorded video signals for display simultaneously on five separate monitors. Two audio channels are also recorded.

f. <u>Integrated GPS</u>. A cockpit-mounted, and integrated GPS is fitted which it is planned will interface with a new Laser Inertial Navigation System that will replace the existing Fin 1010 GEC-Ferranti platform giving a 70 times design reliability improvement.

g. <u>Integrated Defensive Air Sub-System (DASS)</u>. Tornado already has a Radar Homing Warning Receiver and carries both the Boz 107 Chaff and Flare and the Skyshadow ECM pods on the outboard wing pylons. The new integrated DASS suite, which forms part of the Avionics System, is updated by the introduction of the Dash 2 standard Radar Homing and

Warning Receiver (RHWR-2), the upgraded Active Electronic Counter Measure Skyshadow 2 (both of which are supplied separately by the RAF as dependent equipment) and modification of the existing Chaff and Flare Dispenser (BOZ 107). The integration of these Sub-Systems into the overall Avionics System is achieved via a 1553B databus although the Skyshadow retains the Pan standard serial data link with RHWR-2.

h. Nightbird FLIR. A separate Government Supplied Forward Looking Infra-Red (FLIR) system is integrated into the aircraft which comprises a Sensor Head, an Electronics Unit, two Control Panels and a Fairing, Shutter and Window Assembly.

i. NVG Compatible Cockpit. The RAF has already modified a number of its aircraft to be NVG compatible; however, under MLU-93 all displays and control panels in both cockpits will be modified. Provision is made for storage of the NVGs when not in use along with a Power and Built-In Test Unit (PBTU) in both cockpits. The modified lighting system will provide both cockpits with independent selection of two lighting modes, NVG or NORMAL, via a two position toggle switch mounted on the Internal Lights control panel.

j. TIALD 2. As many of you will have seen from news reports during the Gulf War, several of our GR1s were able to carry the TIALD pod and use it with devastating consequences; however, under MLU-93, all GR4 aircraft will be able to carry the latest 400 series pods. TIALD is a pod-based system using thermal imaging, laser and TV to identify, track and designate a target for either self-designation or co-operative attacks, both day and night, and under any weather conditions. A TIALD control panel is fitted in the navigator's cockpit and provides controls for the application of power to the TIALD Pod and for Laser Arming. Data transmission between the Main Computer (MC) and the pod is via the dual redundant 1553B databus.

34. MLU-93 Programme. Turning to the programme itself the MOD has contracted BAe to modify 142 of its aircraft, including all of its GR1A reconnaissance aircraft. The first aircraft entered BAe on 1 Apr 96. This aircraft will be returned to RAF St Athan in Nov 97 in time for the planned GR4 ISD at RAF Bruggen in Apr 98. Thereafter, the ISD for the GR4A at RAF Marham is Jul 98 followed by conversion of the GR1b at RAF Lossiemouth in Apr 2000. The 142nd aircraft is due to be delivered back to the RAF in Nov 2002. At the peak of the programme, BAe will be turning aircraft round in 8 months with a GR4 delivery rate of one every ten days.

35. MLU Support. The programme modifies in total about 160 of the existing LRUs and introduces, excluding certain items of GFE, 9 new major equipments. The new equipments will be supported under an Augmented Logistic Support Contract which will cover the delivery of a new Avionics Ground Training Rig, initial procurement of spare LRUs, production of new AGE and modification to existing AGE, the

provision of publications, the supply of logistic support data pack and ALS for the major new LRUs; the latter will be managed by a 'hole in the wall', serviceable for unserviceable, philosophy. In other words, Industry guarantees to replace immediately any defective item found at the operating base; Industry will then arrange for the defective item to be repaired, tested and re-issued to the front line as a serviceable spare component. In addition, the ALS contract will provide for out of area support, deployment packs and the support of the simulators and rigs. The contract will run for 5 years giving the RAF ample time to define its long term support philosophy based on in-service experience. The contract details pre-determined satisfaction rates with bonus payments for achievements.

36. Post MLU Enhancements. Further enhancements not included in the MLU package, that are being developed and will be embodied either during or after the main programme include: the introduction of a new secure radio, a replacement for the well publicised and criticised IFF system, a new covert Rad Alt, the integration of a new electro-optic reconnaissance pod, a Ground Proximity Warning System, helmet mounted sights and an enhanced Hand on Throttle and Stick (HOTAS) system. We are also looking at expanding the use of JTIDS/MIDS as well as introducing the Conventionally Armed Stand-off Missile (CASOM), the Advanced Air-launched Anti-armour Weapon (AAAW) and the low-level laser guided weapon (Paveway III(UK)). Updates to the Sea Eagle anti-shipping missile and ALARM are also planned. Hence, it can be seen that we are expending enormous effort and money into maintaining the IDS variant through until its planned OSD in 2018. So let me now turn to the ADV.

37. ADV Enhancements. It has long been our objective to enhance the ADV by an MLU-type programme. However, a number of factors have combined to frustrate these efforts. First, as I have already discussed, the aircraft's long-term structural integrity has caused us some concern. Second, there has been a long-running debate as to whether a fighter aircraft, originally designed to counter waves of Warsaw Pact Bombers, with their endurance, large weapon payloads and accompanied by Fighter-escorts, and an aircraft that could never be described as agile, had a long-term future in our inventory. Indeed, many of you will have seen the recent press speculation that we should replace some of our ADVs with F16s. As a result of these uncertainties a number of enhancements have been considered but have suffered progressive financial cuts. However, we have recently gained approval to run a limited enhancement programme approved under Staff Requirement SR(A) 444 and referred to as the ADV Capability Sustainment Programme (CSP). The core programme will allow ADV to carry AMRAAM and ASRAAM; these weapons are already scheduled for procurement for the UK's weapon inventory. In addition to the changes to the aircraft wiring, there are various changes to items of hardware including the fuselage structure and a new missile ejector launcher, as well as changes to a number of avionic components and new software loads. The aircraft will be backward compatible to allow continued carriage and release of Skyflash and Sidewinder. The programme was approved earlier this year. We expect to modify the first aircraft by mid-98 and to receive a formal Military Aircraft Release by October 1998. Thereafter,

we plan to modify 100 aircraft at the RAF's indigenous third line Maintenance Unit at RAF St Athan by 2002.

38. Additional Planned ADV Enhancements. Over and above CSP, we have aspirations to introduce further operational and reliability modifications to the AI24 Foxhunter radar. We expect to introduce the GR4 solid state data entry unit to facilitate a faster loading process for JTIDS. Other enhancements being considered include Helmet-mounted sights, greater thrust from the RB199 Mk 104 engine, a ring laser gyro to replace the FIN 1010 IN platform, an enhancement which should extend MTBFs from 85 to 6000 fg hrs. Finally, we expect to add a Havequick radio capability and to replace the ac's current Mk 10 IFF system.

LESSONS LEARNED

39. So what lessons have been learnt from the Tornado programme to date? First, there is no question that Tornado has been an excellent success story as clearly demonstrated in the pioneering work undertaken in the Gulf War. However, the tri-national collaborative project has not been without its difficulties. There is no question that collaboration has brought the 3 Nations together and produced a versatile aircraft that will remain in front-line service for about 40 years. Moreover, collaboration has had a synergistic effect on the Nations aerospace industries and has allowed the enormous costs associated with design and development to be shouldered by the participating Governments. Likewise, there have been benefits from sharing in-service experience; although, naturally as the individual Air Forces have identified new requirements, commonality of software, systems and weapon carriage has decreased. This is particularly noticeable in the different contents of the various mid-life update programmes. However, there is no question that the collaboration success of Tornado has given EF2000 a sound platform on which to build. Turning to the aircraft structure, the fatigue programmes on the early batch aircraft were unwelcome; as a result, for EF2000, a pre-production fatigue specimen has been stipulated so that any hot spots can be identified and designed out before production commences. Similarly, the structural failures on ADV could have been better managed had we insisted upon pre-production fatigue testing. Conversely, the modular structure of the airframe has meant that even major structural work can be accommodated with relative ease; a benefit that should be read across to new airframes. Looking next at components, equipment qualification is also proving problematical particularly as the 3 Nations are all looking at doubling the aircraft's initial design life. We could have eased this problem significantly by asset tracking so that critical component hours could be recovered by individual equipment serial number, an aspect that is being addressed within the RAF with the advent of the Logistics Information Technology System (LITS). Finally, given the pace of modern technology, combat aircraft must be designed to facilitate upgrades to allow the carriage of new weapons and to permit the integration of new systems. However, I fully recognise that this latter issue will always conflict on new aircraft with demands for reduction in weight, cost and size.

SUMMARY

40. Ladies and Gentleman that brings my presentation to a close. I hope that, by referring to the tri-national

development of Tornado, I have been able to inform you of the very significant work that we have set in hand to ensure the continued airworthiness of Tornado, whilst at the same time planning a number of operational enhancements to ensure that the platform remains effective to fulfil a number of different roles for, in the case of the IDS aircraft, another 22 years. It may seem something of an enigma to talk about Tornado as an ageing aircraft when it has only been in service for 14 years or approximately one third of its planned life. However, I am acutely aware that our fleet leaders have almost reached the end of the aircraft's original planned 4000 hours airframe life and that to extend this life to a figure approaching 10000 hours will keep my engineers, suppliers, financiers and contracts staff busy for another couple of decades

41. I will now be pleased to take any questions that you may have.

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CANADIAN CF-18 STRUCTURAL LIFE MANAGEMENT PROGRAM

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ABSTRACT

The Canadian Forces purchased 135 CF-18 A/B Hornet aircraft in the 1980's. Usage of the CF-18 in the Canadian role was and continues to be substantially different than that defined in the original design requirements. The early usage of the aircraft was very harsh in comparison to design and there were strong indications that airframes would have to be retired before reaching their design service life of 6000 hours. This situation required the adoption of a vigorous and proactive program to manage the structural life of the aircraft. This lecture describes the situation in some detail and then provides descriptions of the programs initiated to gain control over the operational usage and to develop the engineering data that will allow cost effective and safe operation of the aircraft to at least 6000 hours.

INTRODUCTION

The inclusion of the relatively new F/A-18 Hornet (known in the Canadian Forces and referred to in this paper as the CF-18) aircraft in a lecture series on aging aircraft is done for a very practical reason: aircraft can "age" in a variety of ways. One way of looking at the aging processes is to consider three time scales. There is the real time scale which is a simple measure of how old the aircraft is. Another is the usage time scale which is a measure of how far the aircraft has progressed towards its design life. Finally, there is the operational capability scale which is a measure of how capable the aircraft is to perform its role in the current operational environment.

The optimum, from an economic viewpoint, is to ensure that the aircraft proceeds along these scales in a relatively parallel fashion such that the aircraft just reaches its full certification life limits at the time of operational obsolence and at a real time that is maximized.

The goal of the life cycle manager is to ensure this happens. The life cycle manager has several tools in his arsenal that can be used to modify the rate that an aircraft ages along any of these scales. These include repair, overhaul of the airframe at selected times to address time related issues such as corrosion, aggressive fatigue life management programs that modify the rate of usage damage accumulation, mid-life structural modification programs, operational/performance enhancing modification programs and life extension programs.

The Canadian Forces CF-18 situation is one where usage more severe than that anticipated by design was causing premature and accelerated "aging" on the usage scale. This lecture describes what was done to bring this "aging" under control. The lecture will concentrate on the airframe.

DESIGN PHILOSOPHY AND DESIGN SPECTRUM

The CF-18 structural design criteria is based on the United States Navy concepts of fatigue and fracture as reflected in the aircraft detailed specification SD 565-1-1 [1]. This specification essentially follows a safe life approach to certification which requires a test demonstration of twice the planned service life of the aircraft. Compliance was to be demonstrated by fatigue analysis and test using crack initiation as the primary failure criterion.

The basic service life and design requirements are summarized in Tables 1 through 4. These design usage goals were derived by McDonnell Douglas and approved by the United States Navy. The flight manoeuvre design spectrum is based on anticipated usage of the aircraft. This anticipated usage was derived from a review of the proposed role of the aircraft as it would be flown by the United States

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Navy. It is understood that the numerical exceedance values were largely based on data from USN carrier operations. Another version of the F-18 aircraft based on the land based operations of the United States Marines was also considered but ultimately rejected.

One very important design feature of the F/A-18 aircraft was the introduction of active control technology (ACT) coupled with a digital flight control system (DFCS). This adds some complexity to defining its usage since there are additional sensitivities to secondary flight parameters used by the flight control algorithms. The local stress histories at critical areas of the aircraft are not only dependent on the actual manoeuvre flown but also on the speed and altitude (or point-in-the-sky or PITS) at which the manoeuvre is flown. While the pilot command input and the rigid body motion of the aircraft may be essentially identical for the same manoeuvre flown at two disparate PITS, the flight computer decides the most effective control surface deflections to accomplish the manoeuvre. An example is a roll at a low dynamic pressure which involves high aileron deflections while the same manoeuvre at a high dynamic pressure may be accomplished predominantly by differential empennage control movements. The resulting stress/strain distributions are quite different.

An important feature of the USN approach to defining the design spectrum is a conservative approach to defining the PITS to be used for design and analysis. The USN flight manoeuvre design spectrum is based on a combination of four worst case speed/altitude regimes. Each regime represents a critical PITS for a major section of the airframe. The four critical speed altitude regimes correspond to the worst loading, therefore minimum life for the component [2].

To enable fatigue analyses and tests to be performed, the tabular exceedance requirements and the PITS distributions were combined by McDonnell Douglas into a flight-by-flight master event spectrum using the method defined in Reference [3]. For the F-18 A/B design, the result was a randomly generated 300 hour, 250 flight spectrum block which consisted of all the cycles of flight, ground and pressurization loads which had the essential statistical requirements defined in Tables 1 to 4.

Static limits were also defined for the symmetrical and unsymmetrical cases and these are illustrated in Figure 1.

EARLY CANADIAN USAGE SPECTRUM

Overview of Significant Actions Affecting Spectrum Severity

Canada purchased 135 F/A-18 aircraft and began delivery in the early 1980's and the last aircraft was delivered by 1987. Initial usage was confined to basic and operational training and to flight test programs for weapons clearances. The first training squadron was formed in September of 1983 and the first two operational squadrons were formed in June of 1985. The final operational squadron was formed in June of 1989. The operational squadrons were located in Germany, Quebec and Alberta and had significantly different operational roles and operational environments. There were major differences in usage severity amongst the squadrons.

There are distinct phases to the usage severity of the CF-18 that are defined by some significant events. The first significant event was a modification to the Flight Control Computer (8.3.3 PROM) which was implemented on the CF-18 aircraft during 1985. The purpose of this modification was to incorporate a more sophisticated g-limiter system to the aircraft. This essentially allowed a "care-free" manoeuvring attitude to be adopted by the pilots since the fly-by-wire control system would effectively prevent overstress and maximum rate manoeuvres became the norm. This resulted in a significant increase in usage severity.

In March of 1989, as a result of detailed study of the early usage data, the CF implemented a Fatigue Life Management Program or FLMP [4]. The FLMP, which is discussed in a later section, also has affected the rate of damage accumulation on the CF-18 fleet, but in a very positive way.

The aft fuselage and empennage of the CF-18 is prone to buffet induced fatigue damage [4]. The CF-18 is equipped with a leading edge extension (LEX) designed to create a vortex over the inboard section of the wing to delay stall and allow high angle of attack operation [Figure 2]. The strength of the vortex is a function of angle of attack and dynamic pressure. The vertical stabilizers were canted to ensure that they operated in this high energy flow and thereby increase effectiveness.

The result is an induced buffet response of the horizontal and vertical stabilizers that caused a high rate of fatigue damage accumulation. This damage accumulation was reduced substantially by fitment of a LEX-Fence which was essentially a vortex generator that delayed the bursting of the LEX induced vortex. Fitment of this item took place starting in 1988.

CF-18 Usage Monitoring

All CF-18 aircraft are fitted with a Maintenance Signal Data Recording System (MSDRS). The MSDRS was developed by McAir to provide fatigue usage, flight incident records, engine usage data and associated maintenance data. Components of the system comprise an on-board processor and a data recorder that writes to a magnetic tape cartridge. A ground station is used to strip the data from the cartridges and make it available for engineering use.

Various parameters are grouped together in MSDRS messages and identified by record codes. A list of MSDRS Codes pertinent to the development and documentation of fatigue loads and fatigue usage is listed in Table 5. These messages are recorded when triggered by an exceedance of a threshold on selected channels. The fatigue Code 49 is triggered when the normal acceleration reaches a peak or valley and the parameters listed in Table 6 are recorded. Other codes are triggered by engine events or weapons release. The Flight Incident Record (Code 46) provides some flight parameter information and is written as a minimum, every second (Table 7).

The tape cartridge is down-loaded at base level when the tape is full or when there is an accident or incident to be investigated. The raw MSDRS data is transferred to 9 track magnetic tape and a header is assigned to the data. Each block of data containing a header is called an Air Data File (ADF). The ADF header contains pilot identification, mission type for each flight, aircraft tail number, date of ADF creation, mission computer load number, base ID and the airframe hours. One ADF can contain one to ten flights depending on the length and activity of each flight. The ADF's are sent from the bases to a contractor facility for data reduction and individual usage tracking.

Early in the CF-18 program, the ADF files were forwarded to the Naval Air Development Center (NADC) for processing because no facility was available in Canada. CF-18 usage data processing was done in yearly batches and an annual report prepared. The first usage data report was received from NADC in 1984.

Fatigue Life Expended/Fatigue Index

An important concept for understanding the CF-18 usage monitoring is the Fatigue Life Expended Index (FLEI). The FLEI is essentially a measure of the cumulative fatigue damage at the wing root strain gauge of the FS 470 bulkhead of the CF-18. The wing root sensor is affixed to the forward lug of the lower titanium wing root splice fitting . It records strain near the lower edge of the lug neck and corresponds directly with the lateral lug load. The cumulative damage is calculated using a local strain-life crack initiation method normalized to a mean life of 12000 hours with a life reduction factor of 2. An aircraft operated to the design usage spectrum will reach a FLEI of 1.0 at the design service life. An aircraft operated more aggressively will reach this same FLEI in a lessor number of hours. The fatigue life of the aircraft is considered expended at a FLEI of 1.0 regardless of the flying hours.

The FLEI is calculated on an individual tail number basis and was part of the data reduction processing at NADC. In 1987, responsibility for this analysis was transferred to a Canadian contractor. It is important to note that early FLEI calculations were not consistent with the more simple damage calculations done using "g" exceedance data. Several data processing issues, analytical procedures and initial data correction mechanisms have since been developed that allow a more representative FLEI damage to be defined.

The foundation of the FLEI calculation is the USN Structural Appraisal of Fatigue Effects (SAFE) software developed by McDonnell Douglas. The Canadian version, SLMP (Structural Life Management Program) uses a Palmgren-Miner based crack initiation model based on a Neuber local stress-strain approach to determine accrued fatigue damage based on the strains recorded on the wing root MSDRS strain gauge. Detailed fatigue tracking investigations revealed a number of serious deficiencies that resulted in an over-conservative estimate of fatigue accumulation. Some specific issues were:

- *MSDRS strain gauge serviceability assessment*: If a gauge goes "bad", the system moves to a conservative fill-in mode. Manual resetting of the MSDRS is required before the fill-in mode is abandoned. The manual resetting was not done appropriately.

- *Wing root strain drift:* The system used the initialization value of the gauges as a "calibration" of sensor responses. The initialization value only corrects for gauge offset and is not a measure of response differences between aircraft. The wing root gauge in particular goes through a phenomena known as strain gauge drift (which McDonnell Douglas has stated is due to wear on the wing root attachment bushings) which results in an over-estimation of fatigue damage. Calibration factors must be derived for each aircraft based on defined PITS and configurations to correct the data. Data reprocessing is required to correct this error.

There have been some known deficiencies in the current fatigue tracking methodology that has affected its reliability and representativeness. Several engineering studies have been initiated to correct the deficiencies and most importantly (and expensively!), the CF made a decision to re-process all the historical fatigue data. This work is underway and will not be completed until December 1996.

Early Usage Severity

This section will discuss the severity of the Canadian Forces operation before the introduction of a pro-active fatigue life management program.

The first data available to the Canadian Forces on their usage was one summarizing its 1984 usage. The next came in October of 1985. At this time, the usage of the fleet was very immature in that only the training squadron was firmly established. Two operational squadrons had been formed in June of 1985 but their usage had not stabilized. With the 1986 data, it became possible to identify some alarming trends in the usage of the CF-18 in comparison to the design goals and with respect to the usage trends.

The traditional measure of usage severity for a fixed wing aircraft is its "g" exceedance curve and this is an appropriate item to use when evaluating the manoeuvre usage severity of the CF-18 aircraft. Some care must be taken in relating the exceedance curve to damage rates since they do not account for the aircraft weight and configuration nor for the PITS, which is important for the CF-18. Notwithstanding this limitation, Figure 3 taken from Reference 5 dramatically illustrates the trend in usage. Note also the variance of this usage from the design spectrum.

The concern arising from Figure 3 is two-fold. Firstly, the design exceedance curve and the inservice curve are quite different in slope which implies that the cycle by cycle spectra are very different. The implication of this is that the design analyses and tests, which are sensitive to the local stress-strain histories, may not be valid for the CF operation. The design crack initiation analyses, in particular, are based on a local-stress strain approach and are therefore very sensitive to changes in ratios of high load exceedances versus medium load exceedances. It is also noteworthy that typically, most of the damage occurs from the mid range exceedances (~ 4-6 g) rather than the high values because of the much higher number of mid-range occurrences. Therefore, the reduced number of high "g" occurrences in the Canadian operation does not mean a large reduction in damage accumulation.

The other concern is the increasing severity of the Canadian operation. Reference to the damage calculations done at the time indicated an increasing damage accumulation rate of nearly 20% per year, most likely related to the pilots becoming more comfortable and experienced with the aircraft to the point where new manoeuvres and tactics were forthcoming. These damage calculations are suspect because of now known problems with the FLEI calculation, but the trend is consistent with the increased exceedances in 1986 as compared to 1984 and to design (Table 8). Therefore not only was the damage accumulation more severe than anticipated during design, it was increasing rapidly.

The effect of this early severe usage was enhanced by the introduction of a "g" limiter to the CF-18. The effect of the "g" limiter appeared to remove the pilot's concern regarding overstressing the aircraft and ensured that more manoeuvres were conducted at the edge of the flight envelope [6]. Figure 4 illustrates the effect of this change in the Programmable Read-Only Memory (to PROM 8.3.3) on the Canadian Forces operation.

High Angle Buffet Loading

Fatigue cracks were discovered in the FS Y598 vertical fin attachment stub frames of the McDonnell Douglas F4 flight test aircraft in December 1983 after only 736 hours of flying [4]. These cracks were not apparent on the structural test article until 14,902 hours. Similar cracking started appearing in the fleet as early as 300 hours. The conclusion was that the manoeuvre loading was not the main damage mechanism and the suspicion was that high angle of attack buffet was a major contributor.

Extensive McDonnell Douglas flight testing in 1984 and 1985 was done to characterize the buffet environment. The magnitude of the induced loading had been underestimated during design but more predominantly, it was suspected that the percentage of time the aircraft was spending in the buffet damage range had been significantly underestimated.

The intensity of the buffet on the vertical stabilizers is a function of angle of attack (AOA) and dynamic pressure (q). No direct comparison can be made between Canadian inservice AOA/q distributions and design since the design values are not available to the author. The Canadian early experience showed some variation from design for angle of attack frequency of occurrences (Figure 5). The angle of attack comparison does not show that the time spent above AOA 10 degrees was significantly more than the design assumption. The issue of the underestimation of the damage is still relevant to the Canadian operation.

The fitment of the LEX fence reduced the buffet induced damage rate by approximately an order of magnitude which has allowed continued operation in this flight regime.

Mission Profiles

The Canadian Forces missions for the CF-18 were substantially different than the carrier based operations envisioned by the United States Navy. The CF-18 was originally purchased to replace the CF-101 Voodoo interceptor and the CF-104 NATO support fighter-bomber which included both ground attack and interceptor roles. In fact, there were large differences in the mission distributions between the Canadian Forces squadrons and none of these squadron distributions were consistent with the aircraft design assumptions. The result was that the spectrum used for design analyses and test was not representative of CF-18 usage.

This can best be seen by comparing the actual Canadian manoeuvre-PITS distribution with that used for design and test. Table 9 lists the PITS and the percentage manoeuvres used by McDonnell Douglas for design, those flown by the CF-18 during early usage and those contained in the fatigue test spectrum representative of current usage.

As can be seen, very little of the Canadian operation occurs at the design/test PITS. In some ways, this is positive in that the design/test PITS were selected because they are where the most damage occurs for a specific manoeuvre.

Landing Loads

The F/A-18 was designed for operation from an aircraft carrier and as seen from Table 1, the design landing spectrum was very severe. Typically, the USN operation uses unflared landings at relatively high sink speeds (~12 ft/sec). The MSDRS data monitoring system was designed to capture the maximum sink speed in the two seconds before weight on wheels (WOW) and this approach to data capture is totally consistent with the USN landing procedures.

The CF pilots typically use a flared approach. The effect on vertical velocity at touchdown is dramatic as shown in Figure 6 which is from a flight test program. The vertical velocity from the MSDRS system is 9.1 ft/sec whereas the actual vertical velocity was only 2.9 ft/sec.

Because the MSDRS system does not capture the vertical velocity at touch-down, a program was initiated to measure squadron touch-down speeds and compare them to the MSDRS readouts. High speed cameras were used to capture the landings and photogrammetric techniques were used to calculate the actual touch-down vertical velocity. MSDRS data from each measured landing were obtained and compared to the photogrametric results. As shown in Figure 7, the flight test data was confirmed.

The conclusion is that the CF-18 fatigue accrual rate for those areas sensitive to landing loads was significantly less than design.

Discussion - Early Usage

The early Canadian CF-18 usage differed in significant ways to that assumed during design and certification. Specifically, the rate of damage accumulation was more severe and the relative damage accumulation on major components is different because of the differences in manoeuvre-PITS distribution.

Damage predictions showed that if the present damage accumulation rates continued at the then present level, or even more drastically, at an escalating rate, airframes would have to be retired on the basis of expended fatigue life with less than 10 years service. This was unacceptable to the military for both operational and economic reasons.

As a result, two major programs were initiated to recover control of the situation. The first was the implementation of a dedicated and effective CF-18 Fatigue Life Management Program that involved the engineering, maintenance and operator communities in a coordinated attack on usage severity. The second was the initiation of the International Follow-On Structural Test Program [IFOSTP] in collaboration with the Royal Australian Air Force (RAAF).

Another important but lower profile activity that was initiated was the previously discussed review of the MSDRS data acquisition and data analysis methodologies to ensure that the predicted trends are in fact, real.

CF-18 FATIGUE LIFE MANAGEMENT PROGRAM

During 1988, the Canadian Forces proceeded with an integrated approach to fatigue life management. Based on the usage of the CF-18 described previously, the structural integrity managers within the Canadian Forces identified the probability that the design service life of the aircraft would not be achieved unless a fatigue life management program [FLMP] was adopted to reduce the severity of the usage. The program would consist of a coordinated set of activities including:

- a definitive statement of intended operational usage/mission profiles;
- an educational program for aircrew and operational managers on the basics of fatigue accumulation and methods of reducing the rate of accumulation;
- timely and accurate provision of the fatigue status of individual aircraft to fleet managers and operators;
- detailed direction on how this status can be used for individual aircraft management and allocation.

It is worthwhile stating the goal of the FLMP:

"The ultimate goal of the fatigue life management program is to ***monitor*** *and* ***control*** *aircraft usage and the corresponding fatigue damage accumulation such that the* ***economic life*** *of the fleet is maximized while* ***maintaining operational effectiveness.****" [7]*

This commitment of the structural integrity program to maintain the operational effectiveness of the fleet was instrumental in gaining the support of the whole CF-18 community and is the cornerstone of the success of the program. It is also noteworthy that the operator community was involved in the definition of the FLMP.

FLMP requires close collaboration between the operational and the engineering and maintenance communities to ensure maximum availability of aircraft to carry out the assigned tasks. Specifically, the operational community in the Canadian Forces Fighter Group has the responsibility for implementing the FLMP process. The technical responsibility for defining the program rests with the structural integrity group (Directorate of Technical Airworthiness) within the National Defence Headquarters. This essentially means that the operational community controls the process and therefore has direct responsibility for managing the fatigue acquisition rate of individual aircraft at unit level.

For the Canadian Forces, an added complication has arisen because of the harshness of the usage as compared to the design assumptions. Because of this difference, the demonstrated lives and order of failures predicted by the manufacturers certification and compliance demonstration tests may not be representative of those that will be experienced by an aircraft operating to the CF spectrum. For this reason, the CF have adopted a scatter factor of 3 instead of 2 as used by McDonnell Douglas and the USN. This means that the CF currently consider the fatigue life of the CF-18 to be expended when the FLEI reaches a value of 0.667 rather than 1.0 as noted previously.

The underlying approach of the FLMP is to manage the individual aircraft such that the rate of FLEI accumulation and the rate of flying hours accumulation are such that when the FLEI reaches its limit, the airframe has at least the design service hours. For example, based on initial design, FLEI = 1.0 and flying hours are *at least* 6000 hours. This implies an overall average FLEI accumulation rate of 0.167 per 1000 flying hours. For the CF-18 aircraft with the revised FLEI maximum of 0.667, the target average FLEI is 0.111 FLEI/1000 hours. Aircraft accumulating damage at a higher rate than average should be considered for reassignment to a less damaging mission distribution and aircraft with low damage acquisition can be considered for the more aggressive missions. Figure 8 (Ref 5) illustrates the zoning concept for identifying aircraft with a high FLEI rate for its accumulated flying hours.

To assist the operational community to implement this program, the CF have developed some excellent data presentation processes. An example taken from Reference 7 is shown in Figure 9 for an operational squadron within the CF. Note that the individual aircraft status with respect to fatigue accumulation will be affected by the reprocessing of the MSDRS data as described above. However, since the processing errors generally resulted in a conservative prediction of damage accumulation, it is not expected that the fleet will be adversely affected.

The CF-18 historical FLEI rate per 1000 flying hours is shown in Figure 10. This figure is a real indication of the success of the FLMP program. The peak damage accumulation rates have been significantly altered to the point where it is expected that the CF-18 may exceed the 6000 design usage hours with the reduced FLEI target of 0.667. Recent role changes within the CF and the closure of the European based squadrons have affected the damage accumulation rate adversely. However, this early trend of increased severity has been identified and is being addressed by the FLMP.

INTERNATIONAL FOLLOW-ON STRUCTURAL TEST PROGRAM

The International Follow-On Structural Test Program is a collaborative program between the Royal Australian Air Force (RAAF) and the Canadian Forces (CF) and is the most significant item related to the lifing policy and life cycle management of the two fleets. The RAAF purchased 75 aircraft (57 singles and 18 duals) which were delivered between 1985 and 1990. The RAAF experience was similar to the Canadian one whereby their early usage was very severe, particularly for the dual version. This common experience was initially discussed during meetings of The Technical Cooperation Program (TTCP) HTP-3 Structural Integrity Panel meetings from which came a decision in principle to proceed with a collaborative program of testing. Formal discussions were initiated from which a bi-lateral agreement [8] between Australia and Canada was reached to perform a series of full scale tests on the F-18A aircraft.

The agreed objectives of the program are:

- determine the economic life, and in the process, the safe life of the major structural components under a spectrum representative of CF/RAAF operations;
- where possible, obtain crack growth data to support management on a safety by inspection basis;
- validate modifications and repairs;
- establish an engineering data base for life-cycle management through to retirement.

From the Canadian Forces viewpoint, the need for the program was established for the following reasons:

- the CF usage was significantly different than that assumed for design;
- the differences in configuration between the original McDonnell Douglas test articles and the CF aircraft are significant;
- many components had been re-designed and incorporated based on analysis without verification testing;

- the USN approach to certification testing using the worst "point-in-the-sky" approach and a scatter factor of two were not consistent with the CF airworthiness policy which is based on damage tolerance.

Partnership Arrangements

The work share of the IFOSTP program was defined such that there is not any financial obligation on either participant towards the other. The general principle was adopted that the results of the major tests done in the two countries are of equal value in terms of the aircraft fleet management and economic life determination.

Within this concept, the RAAF is responsible for the testing of the aft fuselage and empennage of the aircraft and the CF is responsible for the testing of the forward/center fuselage and wings. The project definition document [8] provided some guidelines that were used to define the detailed test plans and approach to spectrum development. Some highlights of these guidelines are as follows:

- test project must be flexible such that each country could use the results for determining the safe life, economic life and engineering data base relevant to its fleet;

- test structures should be representative of the majority of the RAAF/CF fleet;

- a loading spectrum representing RAAF and CF operational/training usage will be used with combined dynamic and manoeuvre loading where applicable;

- loads to be applied are to be supported by flight test data;

- airworthy repairs will be incorporated and validated by continued testing;

- for safety of flight structure, flaws will be allowed to grow naturally within pre-defined limits.

National project managers were appointed from the CF and RAAF with responsibilities for the administration, supervision and technical coordination of the national projects as well as coordinating the international interactions.

In Australia, the test is being performed at the Aeronautical and Maritime Research Laboratories (AMRL) which is part of the Defence Science and Technology Organization.

In Canada, it was decided that two tests would be performed, one on the center/forward fuselage and another on the wing. This was done primarily because of schedule reasons as there was an immediate requirement to obtain valid test information on the center fuselage and by de-coupling the wing test from the fuselage, the test result would be available sooner.

There are two test agencies participating in the IFOSTP with responsibilities as follows:

National Research Council
Institute for Aerospace Research

- Test Spectrum Development
- Wing Test
- Advanced Repair Development

Bombardier Inc.
Canadair Defence Support Division

- Generation of Balanced Loads
- Center/Forward Fuselage Test
- Engineering Support - Wing Test
- Engineering Dispositions

The Bombardier Inc. work is being done under a contract to the CF. The IAR/NRC work is being done under a inter-government Statement of Agreement whereby IAR/NRC and DND have formed a mutually beneficial partnership which includes a sharing of costs and benefits.

CF-RAAF Spectrum Selection Issues

There were several challenges to be met with respect to the development of the test spectra. Firstly, this was the first attempt at generating a test spectrum based on inservice measured data for an aircraft with a modern digital flight control system based on active control technology. Secondly, although there were similarities in the higher order usage statistics, there were significant differences in the way the RAAF and CF operated the aircraft, particularly in the speed/altitude distributions and store configurations. These differences, because of the sensitivity of the flight control system, resulted in notable negotiations before final compromise spectra were identified. Note that two spectra were required for the aft fuselage test, one based on the high buffet damage period before leading edge extension (LEX) fitment and another for the post-LEX fitment period.

At the time these comparisons were made, basically during the 1987-88 time frame, the usage of either fleet had not stabilized - operational squadrons were just being formed and the spectra were biased towards basic and operational training missions. The general approach to the development of the spectrum was to bias its overall severity towards the most damaging usage which in practice meant the average of the harshest squadron. The use of an "average" spectrum was a significant departure from the USN design approach which strongly biased the design spectrum towards a severe spectrum by making very conservative assumptions with respect to the PITS at which manoeuvres were flown.

Parallel usage data reduction programs were initiated in order to define the CF and RAAF usage that should be considered for the test programs. Canada pursued a vigorous program whereby data from all the operational and training squadrons (4 aircraft each) were analyzed to determine the spectral content on a squadron by squadron basis. This resulted in the identification of the "most severe" squadron and additional aircraft from this squadron were selected for data reduction. The end result was a set of usage statistics that represented the average of the harshest CF squadron. Note that this process was followed twice: once for the pre-LEX period and again for the post-LEX period.

The data from each of the aircraft in the severe squadron was scrutinized for quality. Because of the large number of data channels necessary for the loads process, it was difficult to identify a single aircraft with enough data and with the necessary variety of missions to use for the data block. In the end, four aircraft were selected for both the pre-LEX and post-LEX mission blocks. Particular attention was paid to the serviceability and response of the empennage strain gauges because the Australian loads process used these strain gauge readings directly in their process. The Bombardier Inc. developed loads process was parameter based and did not use the strain data directly although the data were used for verification purposes.

The Australians followed a different path but reached the same endpoint - pre-LEX and post-LEX data blocks representative of their usage. Their loads process for the empennage structure made extensive use of the four aft fuselage strain gauges.

Hewitt et al. [9] provided an excellent discussion of the issues involved in loading spectrum determination for the CF-18 and Noll [10] discusses the impact of active control technology on structural integrity.

Time History Characterization

The loads process that was followed required a representative MSDRS flight by flight data block arranged sequentially to provide a representative time history of aircraft usage. Organized in this manner, the data block would essentially provide a time history of the major parameters from which the master event sequence could be generated. For the IFOSTP, a data block of approximately 300 flights was selected which is essentially representative of an annual cycle of flying. Figures 11 and 12 provide an indication of the target post-LEX mission history and the actual mission history of the test spectrum for the two most damaging mission types. This close agreement is also apparent on the other mission types. Target parameter statistics were also met with very good accuracy. Figure 13 shows how closely the g spectrum was matched.

The time history for the Canadian center fuselage spectrum was characterized in terms of manoeuvre and PITS using a manoeuvre identification program [9]. This program first eliminated all periods of inactive flying by identifying time slices when the roll rate was near zero, the angle of attack (AOA) was below 10 degrees and the normal acceleration was approximately 1. All other time slices were identified as either a standard manoeuvre (a turn, pull, push, rolling pull or roll) or a non standard manoeuvre (an extended pull, AOA excursion, roll and pull, roll with g, roll then pull or a pull then roll). This was achieved by testing for g ranges, calculating roll-through angles and noting roll directions and the sequence of roll rate and g peaks and valleys and then comparing the observed data against pre-defined manoeuvre characteristics. The end result was an ordered list of manoeuvres to be simulated on the center fuselage test.

Ground Load Sequences

Based on a comparative damage approach, it was determined that the only significant ground based loads that needed to be considered for the IFOSTP test spectrum were the landing events and cycling of the main landing gear.

Landing Events

A landing loads spectrum was developed by first obtaining a relationship between the actual sink speed at touchdown, as recorded by a photogrammetric survey, and the MSDRS recorded sink speed. This was necessary since the MSDRS recorded sink speed occurs about two seconds prior to touchdown and may be significantly higher than sink speed at touchdown for flared landings. Analysis of MSDRS data for the selected squadron and the actual sink speed to MSDRS sink speed correlation obtained from this study were then used to define a distribution of sink speeds for the spectrum. Typical sequences obtained from a landing loads survey conducted by the Aerospace Engineering Test Establishment (AETE) which corresponded to selected ranges of sink speeds were then used to develop the test sequence loads.

Gear Cycle Frequencies

Cycling of the MLG during maintenance was determined to be the only other ground load event of significance to the fatigue life of the centre fuselage structure. The fleet statistics were matched in the block.

Loads Derivation Methodologies

Original Loads Derivation Methodology

The manoeuvre identification process resulted in a very large number of manoeuvres. The original intent was to group these manoeuvres into bins defined in terms of flight parameters and control law boundaries where it would be reasonable to assume that the loads for all manoeuvres within a bin were either constant or could be simply extrapolated. The loads at each turning point of a representative manoeuvre within a bin would then be calculated from a combination of the MSDRS and measured flight data, computational fluid dynamics models and wind tunnel data.

Revised Loads Derivation Methodology

Because of the inadequacies of the analysis tools for predicting loads in high angle of attack aircraft, increasing use had to be made of flight measured loads. An empirical Parametric Loads Formulation (PLF) was developed by Bombardier Inc. based on a knowledge of the aerodynamic loading actions and an analysis of the flight loads data that gave the loads as a function of flight parameters, control surface deflections and some strains.

Since the process was quite rapid and the MSDRS records strains and flight parameters at every significant turning point of most of the strain gauges as well as turning points for 'g', loads could therefore be calculated for every significant turning point for the centre fuselage. The only exception to this was for some symmetric, high wing torque, low wing root bending moment cases where the torque peak was not quite coincident with any strain trigger.

Binning was therefore not required and the manoeuvre identification was only used to eliminate periods of inactive flying.

Data Processing Problems

Significant effort was required to validate and correct data that was input manually. Examples were found of incorrect mission codes, improper stores codes, missing or incorrect flight dates, wrong tail numbers as well as flight time discrepancies. It is therefore apparent that any monitoring systems should be fully automated and that manual data entry must be avoided. Data validation needs to be extensive and performed as early in the process as possible. It is much easier to correct errors prior to major processing.

The normal validation problems were further exacerbated by the very large amounts of data involved. It was very difficult to check data and difficult to check all assembly processes. Files were often so large that they had to be split before they could even be interrogated. It is therefore suggested that considerable effort be devoted to devising automated checking routines that are integrated with the process at every step.

Data Deficiencies and Inaccuracies

Control Surface Deflections

Control surface deflections are measured only once every 5 seconds on the MSDRS system. Therefore, intermediate values had to be derived using the once per second flight parameter data (AOA, Mach number, dynamic pressure, roll pitch and yaw position, rates and accelerations, lateral and normal accelerations and stick position) and the flight computer control laws.

Roll Acceleration

Roll acceleration was not recorded on the MSDRS and had to be derived from the once per second roll rate data. For short duration manoeuvres with very high rates of change of roll angle, this lead to significant inaccuracies.

Rudder Deflections

Poor prediction of the rudder deflection was found to be caused by inaccuracies associated with the stick input. The PLF was therefore reformulated in terms of differential horizontal stabilator strain which can be obtained from the stabilator strain gauges. The re-formulation improved the accuracy.

Trailing Edge Flap Deflection

Errors in predicted trailing edge flap deflection were found to be caused by a difference between the flight measured AOA and the MSDRS measured AOA which is the pilot's Heads Up Display (HUD) AOA. To improve the presentation to the pilot, the HUD AOA is discretized into bands and always rounded up to the nearest value. In addition, the HUD AOA is filtered using two different time constants to improve the readability for the pilot. A comparison of the flight measured AOA and the HUD AOA is shown in Figure 14 together with a corrected HUD AOA based on knowledge of the time constants and the rounding process. Using the corrected AOA improved the trailing edge flap deflection prediction considerably. With the improved trailing edge flap deflections, the wing root torque prediction was generally quite accurate. An example is shown in Figure 15: the points noted as residuals are those points that remain after truncation and re-assembly of stress sequences for a number of critical locations and are those points that will be applied to the test article.

N_Z Discrepancies

Investigation of the difference between the flight measured and MSDRS N_Z showed that it was due to a time lag in the MSDRS system of about 0.1 seconds. A comparison of the flight measured N_Z and the MSDRS N_Z is shown in Figure 16 together with a corrected MSDRS N_Z.

Implications for the Center Fuselage Test Calculated Loads

The calculation of the loads for the center fuselage test were based on the determination of balanced conditions for the major interface loads at the wing root, center fuselage and aft fuselage. Load lines were selected that corresponded to load reversals at these major interfaces thereby ensuring that all major load reversals which could cause fatigue damage to the fuselage structure were included. Interface loads were also calculated at the wing/control surface interfaces in order to ensure proper aerodynamic distributions on the wing.

Calculated loads were verified by comparing against flight measured loads for typical missions that were not part of the data set used to derive the PLF coefficients. When control surface positions and flight parameters measured directly during the flight loads survey were used with the PLF method, the predicted loads agreed very closely with the flight measured loads. However, use of the MSDRS data resulted in some significant errors in wing root torque due to inaccurate rudder deflection and trailing edge flap deflection predictions using the MSDRS inputs. The predicted wing root bending moments were also less accurate when using MSDRS inputs because of differences in MSDRS and flight measured normal acceleration, N_Z.

A comparison of measured and predicted wing root bending moment is shown in Figure 17: the agreement is excellent.

Implications For Wing Spectrum Development and Loads

The same MSDRS data blocks were used for derivation of the load spectrum for the wing test, only the interface loads used for the selection of load lines to be included in the test were based on the wing load reversals rather than those of the center fuselage. In addition, dynamic loading of the outer wing and of the leading edge and trailing edge flaps has been identified as potentially damaging therefore the wing test spectrum must also address this loading.

While the noted inaccuracies and uncertainties in the control surface deflections do not have a significant effect on the center fuselage interface loads derived using the PLF method, they do have a large influence on the accuracy of the wing load distributions. This was not critical for the centre fuselage spectrum since the primary objective was to get the main interface loads at the wing root and specific fore and aft fuselage stations correct. However, for the wing test, the loads must be derived using essentially the same data base as was available for the centre fuselage test. The control surface deflections therefore become more important and alternative methods may be required.

Two approaches were considered. The first was to use a predictor/corrector method, based on the latest F/A-18 six degree of freedom model, to generate parameters at a higher frequency than that recorded by the MSDRS. Some parameters additional to the MSDRS set need to be generated (e.g. roll acceleration, control surface positions). The enhanced MSDRS parameters so obtained can be used as input to the PLF method. This is a more complex version of the current method.

The second method considered and subsequently rejected, was to use the manoeuvre identification program to characterize all manoeuvres in both the required test spectrum and all existing flight test data where flight parameters and control surface deflections were recorded continuously (but not necessarily loads). The concept was to find equivalent manoeuvres in the flight test data to those in the test spectrum and calculate loads and distributions for the former since the parameters are known more accurately. The major difficulty was to define equivalent manoeuvres but practically, the approach failed because there was insufficient flight test data to cover even the most frequent manoeuvres.

CENTER FUSELAGE TEST

The CF-18 centre fuselage test is shown in Figure 18. A total of 64 hydraulic actuators, the two production Main Landing Gear (MLG) retract actuators and six static reaction points are used to load and restrain the aircraft.

For each of the load conditions identified by NRC/IAR for inclusion in the spectrum, actuator loads were derived through an optimization process to best match the calculated loads over the test section. Typically, wing root shear, bending moment and torque and fuselage vertical bending moments are matched to within less than 1.5% error.

Active testing is in progress and the test specimen at the time of preparation of this paper (May 1996) has approximately 6000 test hours. The test has encountered 3 significant events: the first one occurred at 4,712 sfh with a series of failures on the LHS wing fold lug sets, and on the control surfaces (LHS ILEF front spar transmission and RHS TEF inboard hinge). These are on transition structure where the loading is not fully representative. The second significant event was the initiation and growth of a crack in the number 2 fuel tank skin and the third is a discovery of a crack at the centreline of the Y470 bulkhead. These are both areas which were known to be problematic from the certification test. Bonded boron patch repairs are being investigated for bulkhead repairs.

AFT FUSELAGE TEST [3]

As discussed previously, the CF-18 is equipped with a LEX designed to create a vortex over the inboard section of the wing to inhibit separation and delay stall (Figure 19). The vertical fins were canted to take advantage of this high energy to increase effectiveness. The result is that during medium to high angle of attack manoeuvres, severe buffet of the empennage surfaces occur.

There is a synergistic interaction between the quasi-static manoeuvre loading and the higher frequency buffet loading with respect to fatigue damage. The general affect is that the buffet cycles are applied at high mean loads which increases their contribution to fatigue damage. This phenomena is well understood and some attempts have been made by McDonnell Douglas to perform representative (i.e. correct mean plus buffet) loads on the CF-18 compliance tests [3]. However, the loads were not applied realistically in terms of frequency but rather as calculated resultant loads at the normal fatigue test rates.

The Australian approach has been to simulate as realistically as possible, the loading conditions representative of actual flight conditions. For the IFOSTP program, a decision was made to generate the actual modal vibrations at the correct frequencies and simultaneously, apply the manoeuvre loading. They entered into major development programs to develop the loading methods and equally important, the control systems. The essence of the load application system is a unique rolling sleeve pneumatic actuator that has a soft spring stiffness. Using this system, the distributed manoeuvre loads can be applied without affecting the effective stiffness of the empennage component. Concurrently, electromagnetic shakers apply the dynamic loading. Combined closed loop operation of the airsprings and hydraulic shakers has been successfully developed.

The final test arrangement is shown in Figure 19. Several opposing airsprings are used on each empennage surface to allow bi-directional loading. Thrust loading, engine 'g" loading, empennage drag loading and fuselage side loading are also applied in a time coordinated fashion. The overall concept was applied to the ST-01 test article which was loaned to AMRL by the United States Navy.

At the time of writing (May 1996), active testing is in its early stages at the AMRL

WING TEST ACTIVITIES

The wing test activities are well underway at the IAR/NRC with the technical support of BI/CDSD. A technical review has been completed that provided firm direction on several issues including configuration of the test article, loading actions to be considered, and the areas of the wing likely to experience fatigue. Based on an assessment of available failure information, 108 locations were highlighted of which, forty six were prioritized and selected for further investigation as test control locations for spectrum comparison and monitoring. Twelve have been confirmed as the test control points. The technical review also involved the assessment of the available options for the derivation of both manoeuvre and dynamic loads.

The derivation of the wing manoeuvre load distributions is complicated by the digital flight control system of the aircraft which allows variation of the control laws with speed and altitude. This situation, and the fact that there are large control surfaces on the wing, has necessitated investigations into load distribution methodologies. A review of dynamic loads data available from the RAAF Aircraft Research and Development Unit (ARDU) flight tests concluded that dynamic activity is severe enough to warrant further investigation for its applicability to the wing test. Additional ARDU flight tests are in progress to support both the manoeuvre and dynamic loads derivation efforts.

The test rig design is progressing. A retired US Navy F/A-18 center fuselage will be used as reaction structure to ensure proper distribution of internal stresses at the wing root. All major wing components have been received by NRC/IAR and have been inspected to verify integrity and proper configuration. The tasks completed to date include the definition of the test article configuration and the comparison with CF and RAAF fleet effectivities, the definition of the flight test instrumentation and ground calibration requirements and the preliminary definition of load cases for whiffle tree design. Rig assembly has been initiated.

The test is scheduled to start in late 1998.

STRUCTURAL INTEGRITY RELATED PROGRAMS

In addition to the IFOSTP program and the FLMP program, the CF is pursuing a number of structural integrity related programs including:

- review, certification and incorporation of engineering change proposals from McDonnel Douglas that improve upon the fatigue performance of the airframe;

- an Aircraft Sampling Inspection (ASI) that will provide a mid-life health check of the airframe and systems and will be particularly valuable for evaluating the extent of corrosion damage;

- a rationalization of life limited item maintenance requirements with the aim of optimizing airframe inspections;

- an extensive review of the unique operational, maintenance and engineering efforts required to support the eleven early production aircraft that do not contain a number of the life enhancing production retrofits;

- investigations to support the Composite Repair Engineering Development Program (CREDP), an effort to improve upon and expand the composite material structural repair manuals; and

- an effort to quantify and subsequently certify the life improvement factors associated with shot peening. This technique has been used extensively to improve upon the fatigue performance of the structure; however, lifing predictions can presently place only little value in the benefits.

LESSON'S LEARNED

Going back to one of the initial assertions in this paper, the optimum situation from an economic viewpoint, is to ensure that the aircraft proceeds along the aging scales of real time degradation, usage accumulation and operational effectiveness in a relatively parallel fashion.

The major lesson learned from the CF-18 situation is that the "ageing" process starts on day one of entry to service, particularly for the usage accumulation. It is imperative that management of fleet usage be initiated concurrent with introduction to service. The implication of this is that the infrastructure in terms of usage monitoring, data handling, mission definitions, timely data reduction and rapid data dissemination is considered during the acquisition and pre-delivery phases of the new aircraft program.

Another important lesson for both the design community and for the inservice life cycle management community is that the aircrew will do their job and develop flying techniques that give them an advantage in a combat situation. They will take full advantage of the operational capability of the aircraft. This trueism is particularly relevant when there has been a technology leap between the new aircraft and the one it is replacing. It is not conservative to use the usage statistics from a previous generation aircraft to set the design goals of a new generation aircraft without serious consideration of the operational impact of any new technology.

In terms of the CF-18, there were two instances of where the operational community developed tactics which utilized the capability of the active control/digital flight control system and had a detrimental affect on fatigue damage rates. These were the use of the "care-free" manoeuvring capability to maximize performance and a much higher than anticipated use of the high angle of attack regimes during air combat manoeuvring. The former increased the severity of the 'g' exceedence counts while the latter increased the time that the empennage and engine operated in the buffet regime.

Some specific lessons learned:

- Multi-channel computer based data monitoring systems are vital to the development of relevant usage statistics, whether retro-fitted or initially fitted. At an early stage in the definition of the monitoring system, the approach to data reduction and presentation must be considered. Reference to this data reduction approach will assist in the determination of the number and type of data channels, the frequency of recording and the determination of the data collation and storage formats. It will also be cost efficient.

- All monitoring systems should be fully automated and manual data entry must be avoided. Data validation needs to be extensive and performed as early in the process as possible. Because of the large amounts of data that must be handled, considerable effort must be devoted to devising automated checking routines that become integrated with the process at every step.

- It is vital that any fatigue life managment program involve the operational community if it is to be effective. This involvement must occur at all stages from initial goal setting of the program through definition of policies and procedures to final implementation.

- Initial testing of an aircraft is aimed at demonstrating compliance and the engineering data generated is not always useful in defining inservice engineering actions. The design usage assumptions must continually be compared to the in-service usage actualities to ensure relevance.

Finally, it is never too late or *too early* to become proactive in managing the ageing of an aircraft.

ACKNOWLEGEMENT

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TABLES

Flight Hours	6000
Ground-air-Ground Cycles	5000
Field Taxi Runs	3500
Catapult Launchings	2000
Landings	
Field Landing Practice	3000
Field Carrier Practice	3000
Arrested Landings	2000
Carrier Touch & Go	300

Table 1. Design Service Lives, SD 565-1

% of Manoeuvres	Mach No.	Altitude
50	0.8	Sea level
45	0.95	15000 ft
5	V_L	Sea level

Table 2. Critical Points in the Sky for Manoeuvre load determination

Positive (g exceedances)	Cycles/1000 hrs
3g	12000
4g	5700
5g	2500
6g	920
7g	320
8g	100
9g	27
Negative (g excedances)	
0g	600
-0.5g	118
-1g	70
-1.5g	35
-2.g	5.4

Table 3. Exceedances of Manoeuvre Loads

Normal Load Factor	% Asymmetric Manoeuvres
3.5	31
4.5	42
5.5	32
6.5	27
7.5	17

Table 4. Asymmetric Manoeuvre Distribution

MSDRS Record Code	Description
4	Fatigue monitoring - weapons inventory
21	Recorder initialization
22	Recorder summary message
31	Engine data life cycles
46	Flight Incident records
47	Fatigue - landing
48	Fatigue monitoring - initialization
49 to 62	Fatigue sensor peaks and valleys
65	Configuration message

Table 5. Primary MSDRS Codes for Usage Monitoring

Parameter Measured	Frequency (HZ)
IAS	1
Pressure altitude	1
Roll rate	1
Angle of attack	1
Longitudinal stick position	1
Lateral stick position	1
Rudder pedal position	1
Normal acceleration	1
Fuel quantity	0.2
Control surface positions	0.2

Table 6. Flight Incident Parameter List

"g" Level	Percent Increases for 1986	
	Over 1984	Over Design
4	43	96
5	65	58
6	200	33
7	233	-38

Table 7. Percentage of "g" Exceedances for 1986 compared to 1984 and Design (Reference 4)

PITS	DESIGN (FT-01)	TEST ST-16	1985 CF	IFOSTP TEST
V_L, Sea Level	10%	5%	0.1%	
Mach 1.0, 15000 ft	10%	45%	6.2%	
Mach 1.2, 15,000 ft	10%	0	0.2%	
Mach 0.9, Sea Level	70%	50%	5.2%	
Mach 0.8, Sea Level	0	0	12.3%	
Mach 0.9, 15,000 ft	0	0	27.5%	
Mach 0.8, 15,000 ft	0	0	15.5%	

Table 8. Distribution of Manoeuvres by PITS

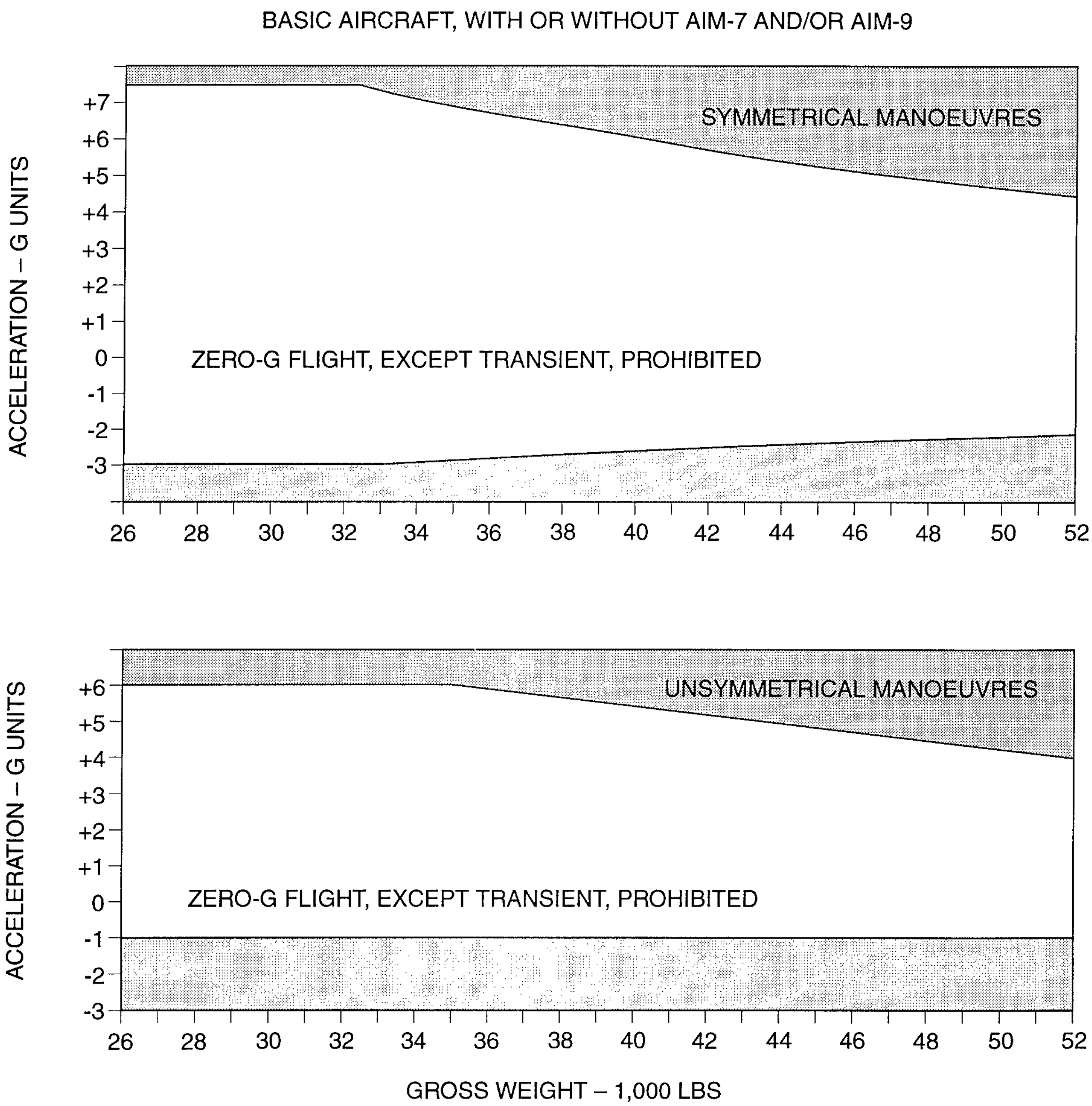


Figure 1. Acceleration Limitations

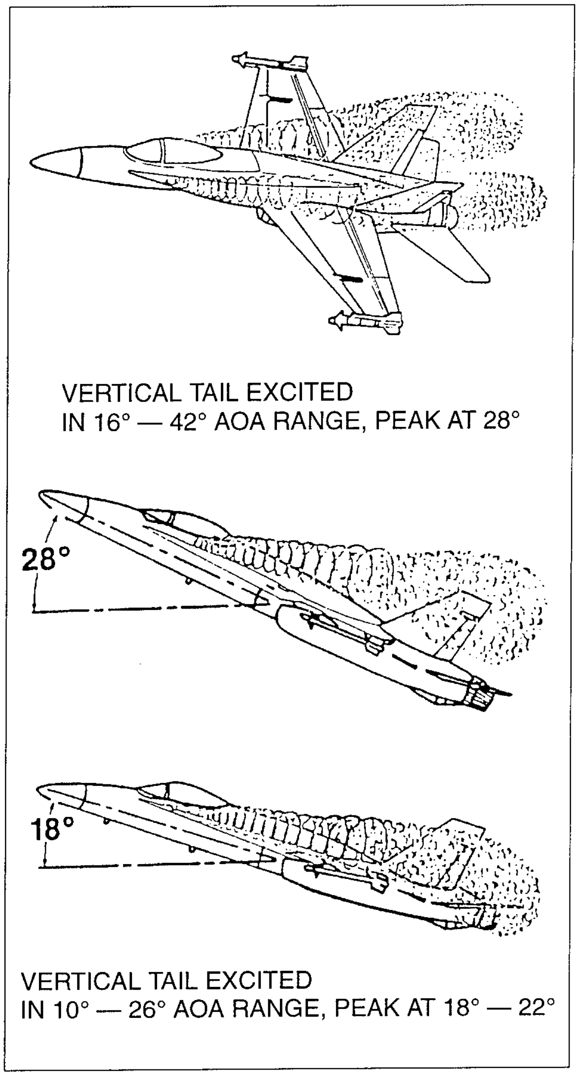


Figure 2. LEX Vortex Excitation of the F/A-18 Empennage

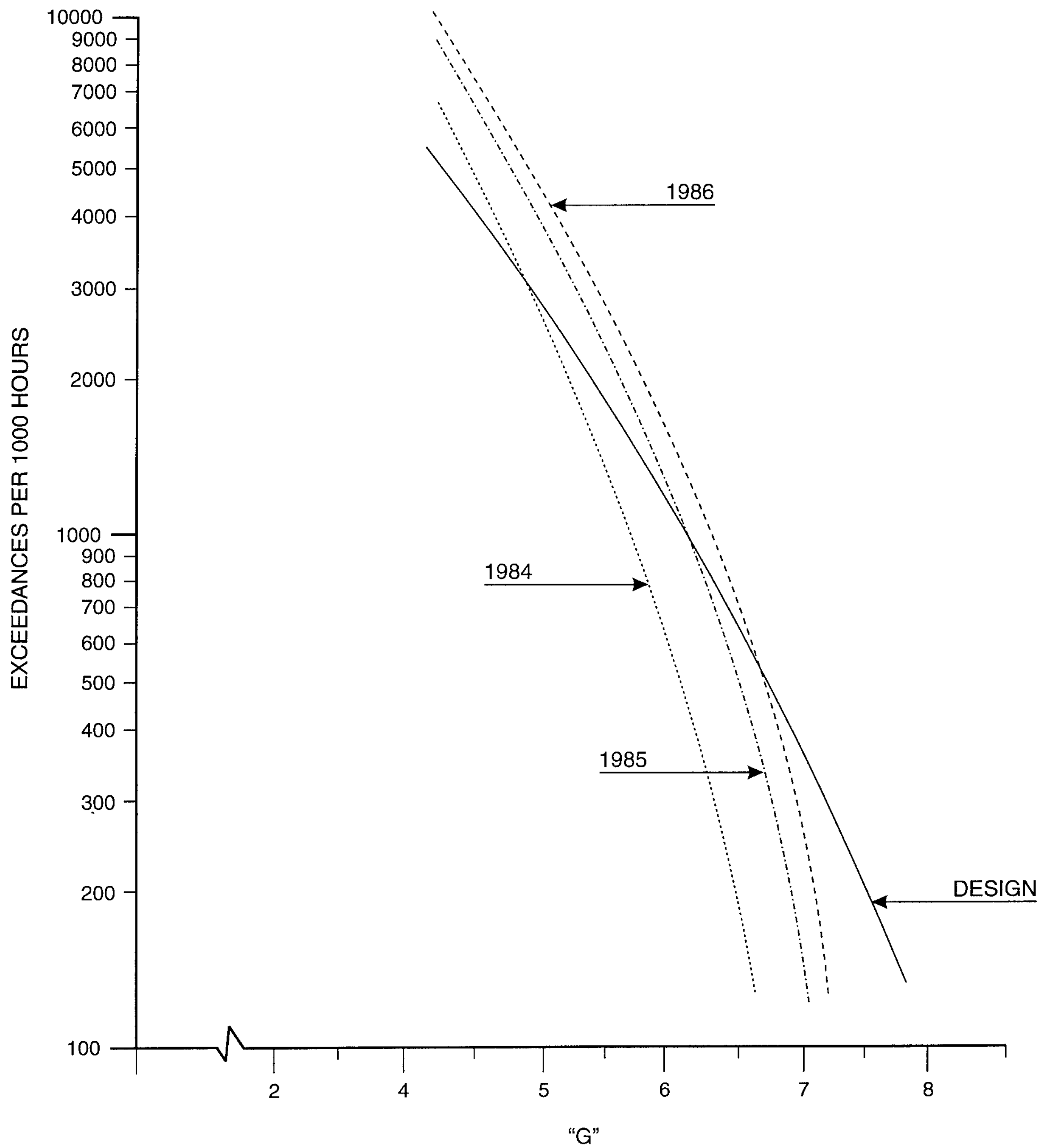


Figure 3. "G" Exceedences through 1986

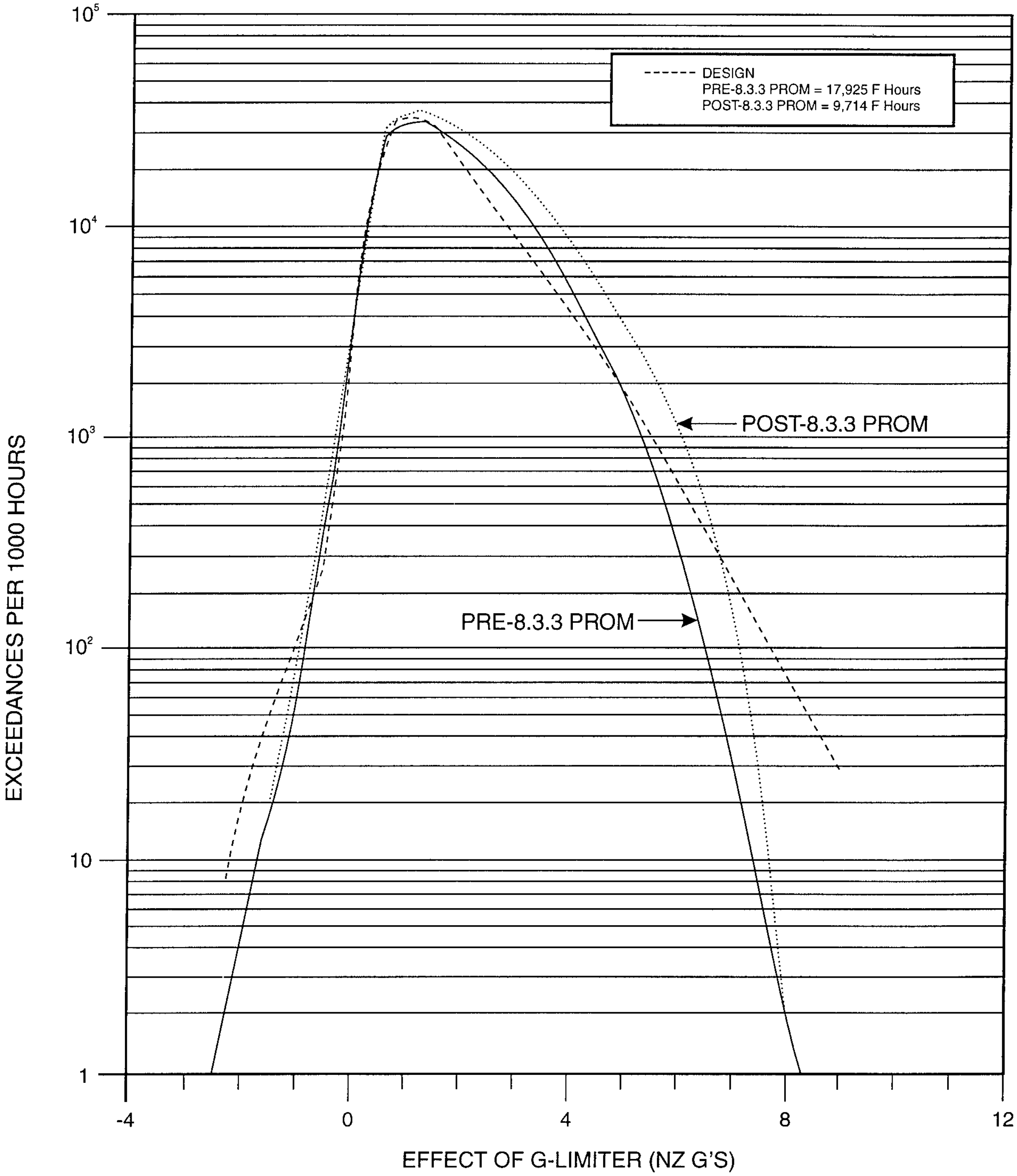


Figure 4. Post/Pre-833 PROMS NZ Usage

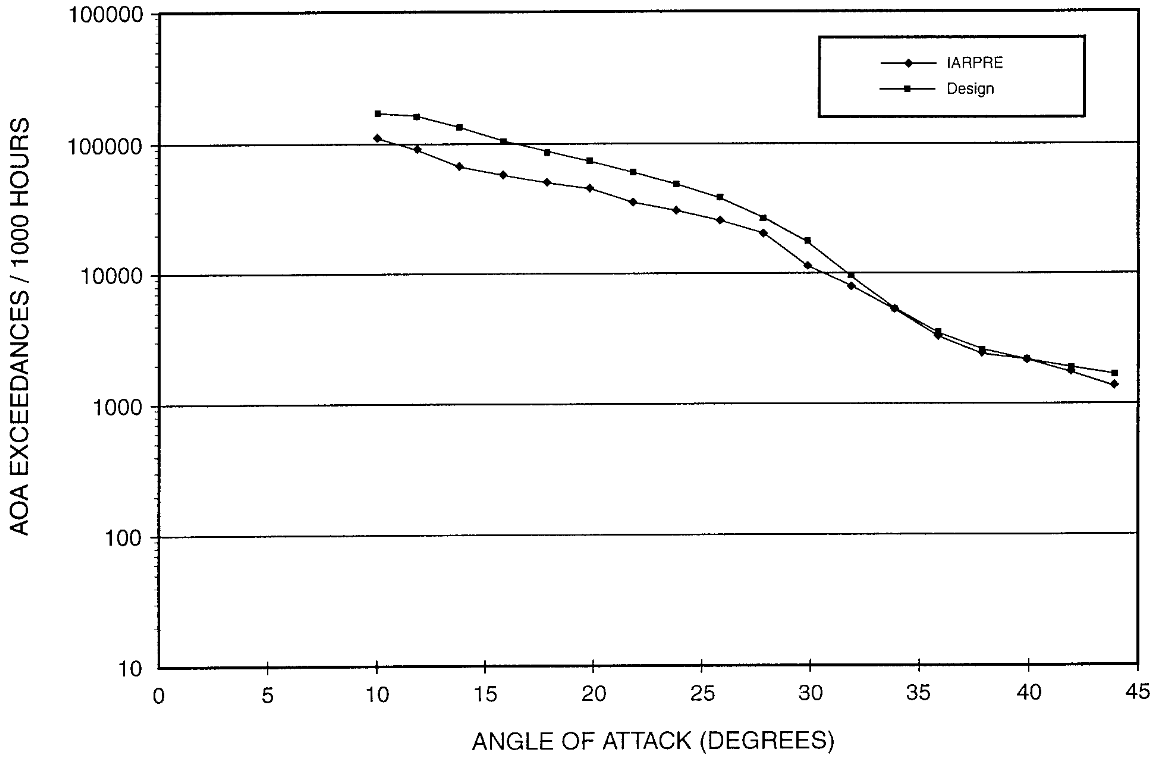


Figure 5. AOA Exceedances Curves for McAir Design and IARPRE Mission Block

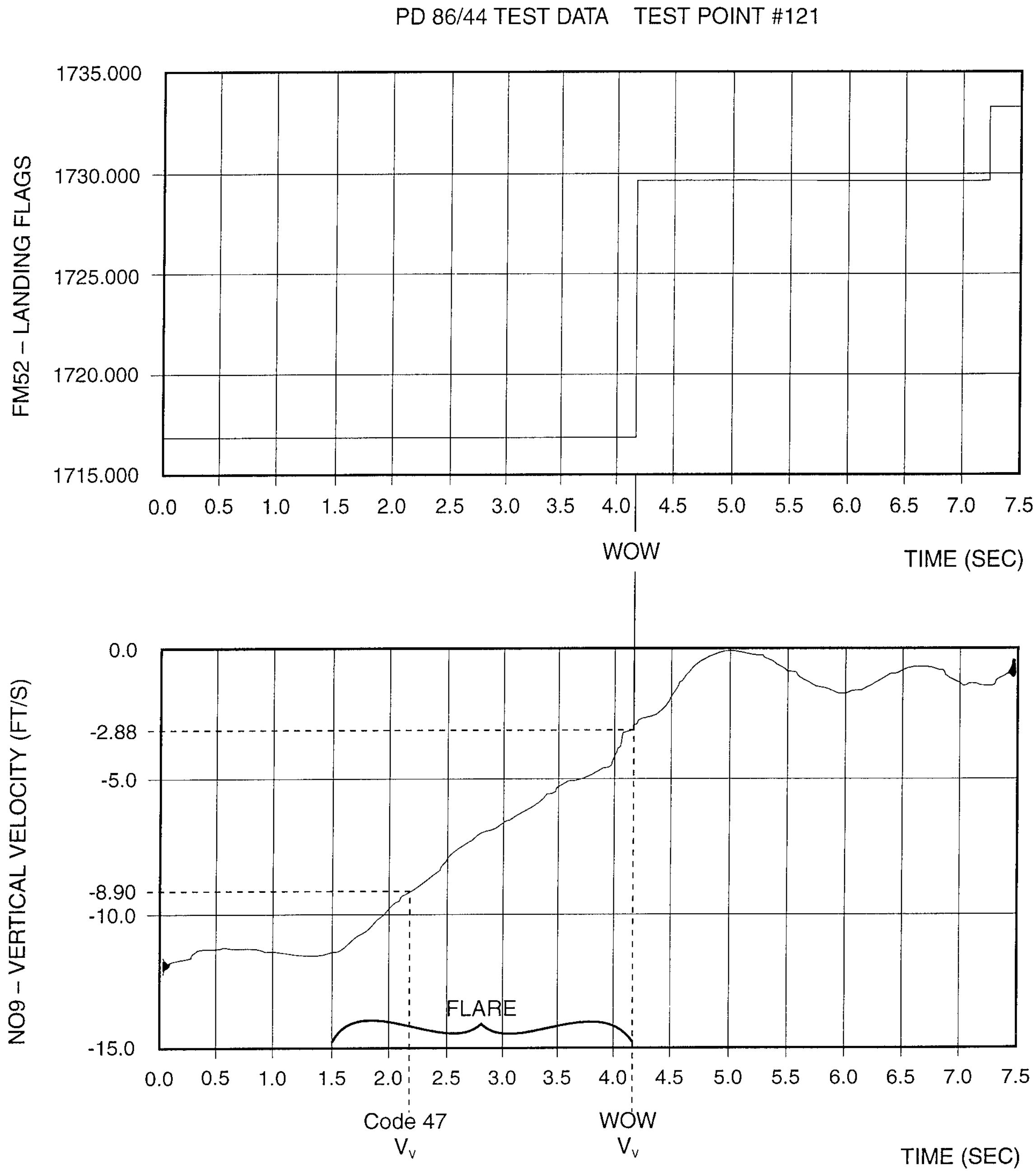


Figure 6. Sample pcm Data Trace from PD 86/44

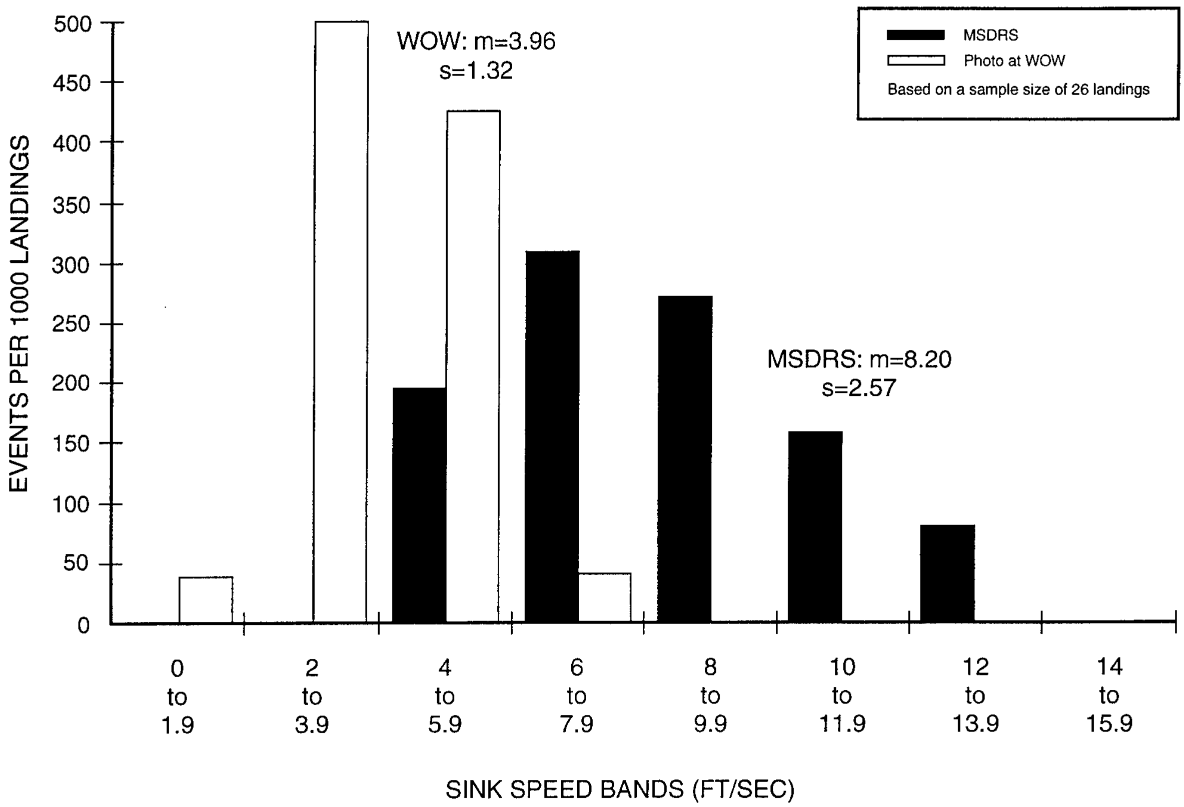


Figure 7. Vertical Velocity Distributions: MSDRS vs Photo

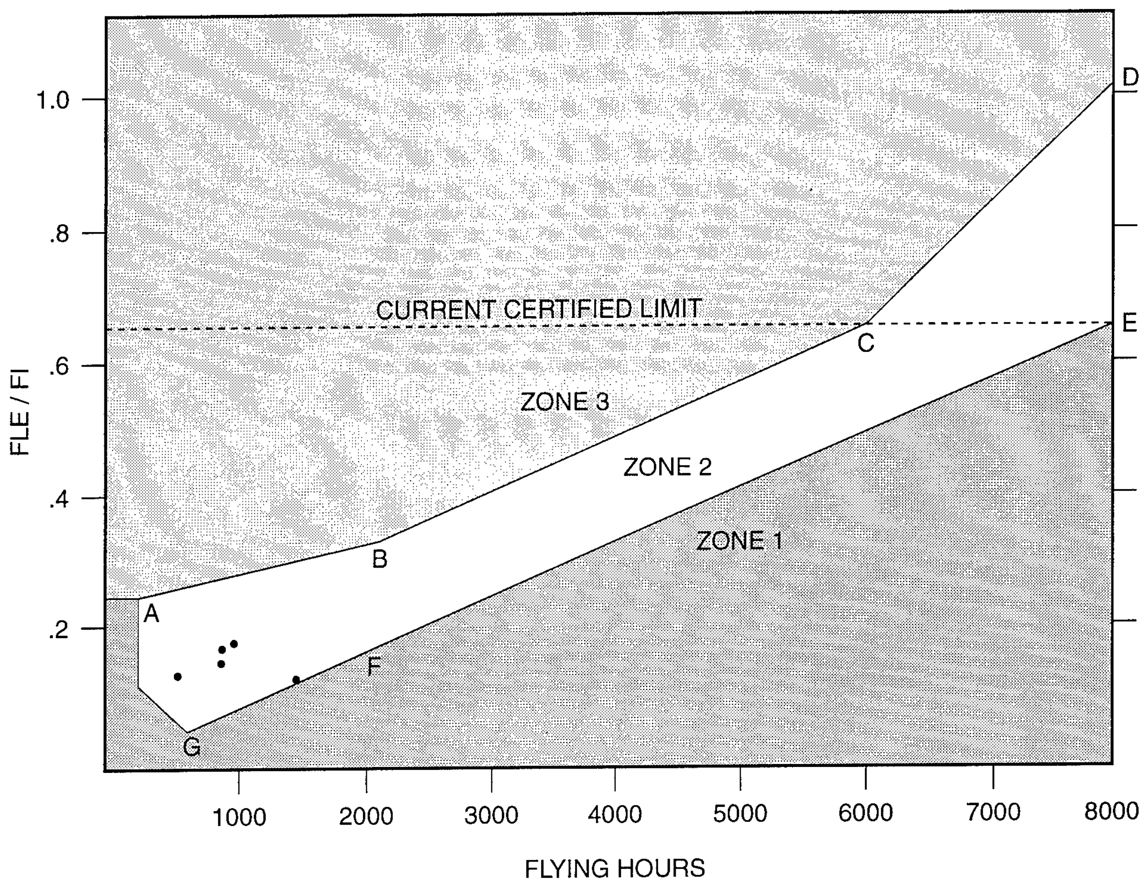


Figure 8. Fatigue Management Envelope

Figure 9. Fleet Monthly FLEI Rate Bar Chart

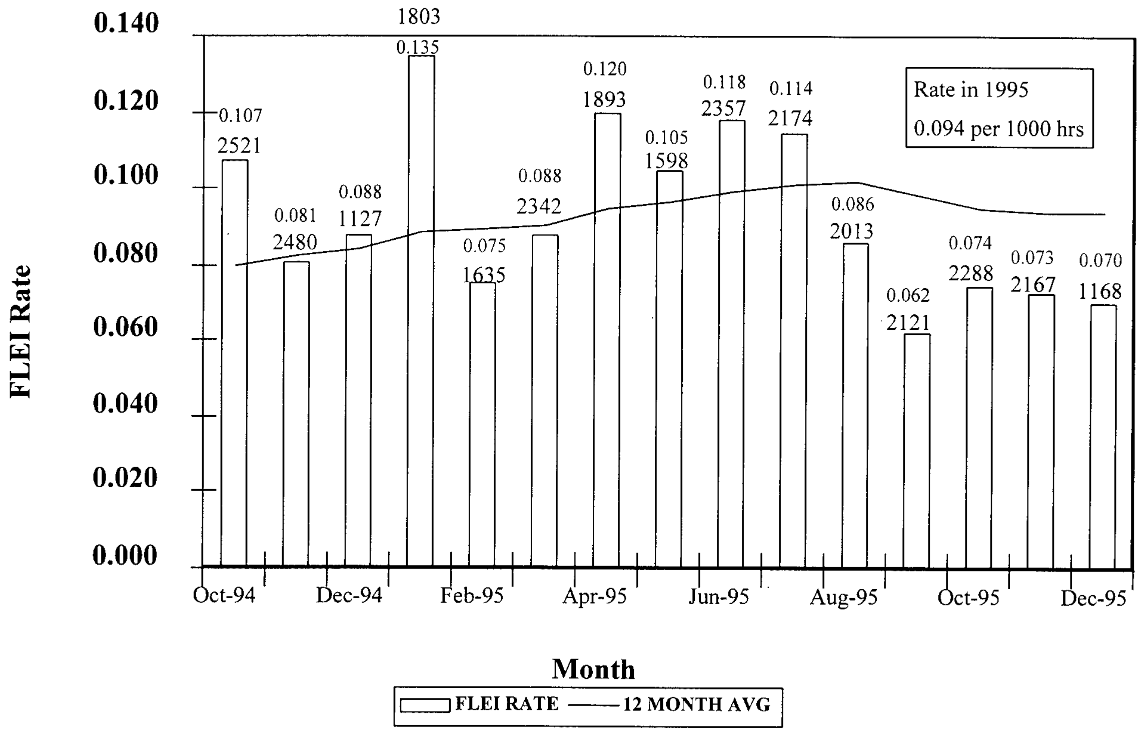
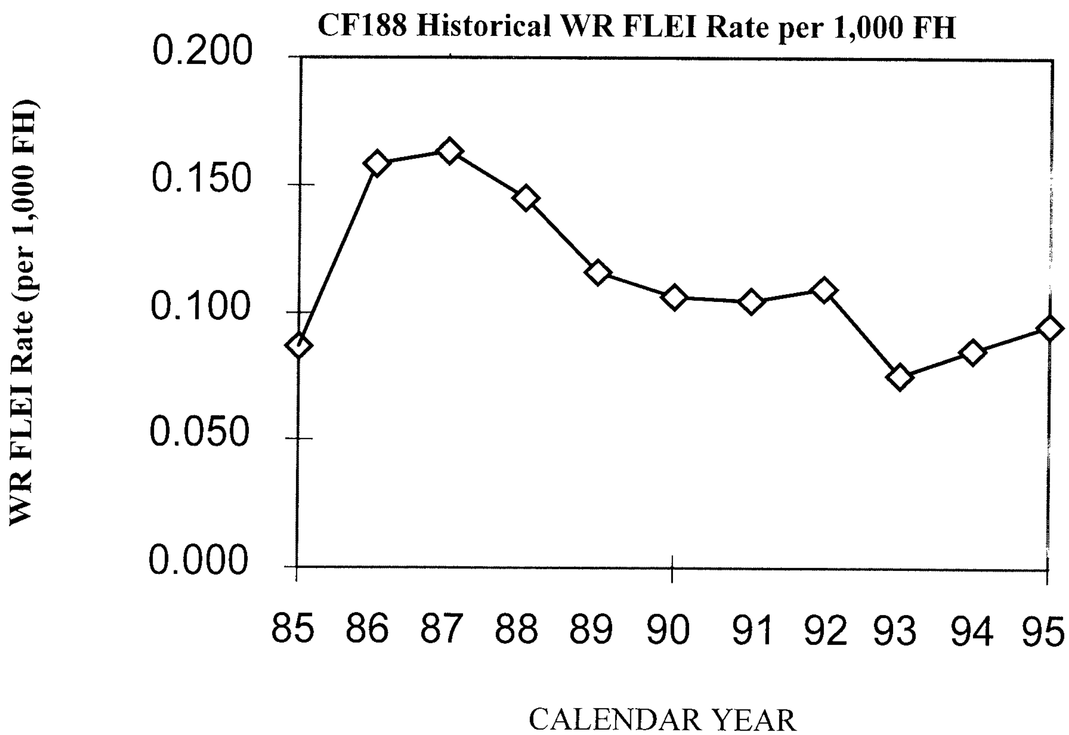


Figure 10. CF18 Historical Structural Fatigue Consumption



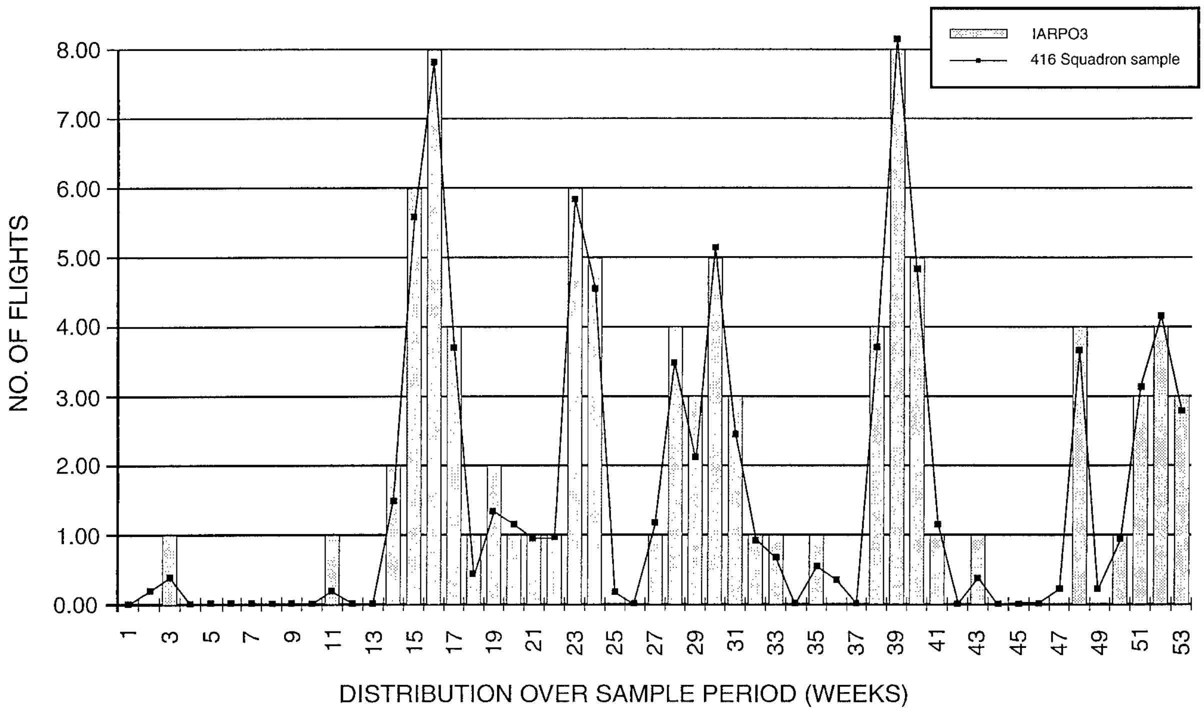


Figure 11. Mission D Sequence

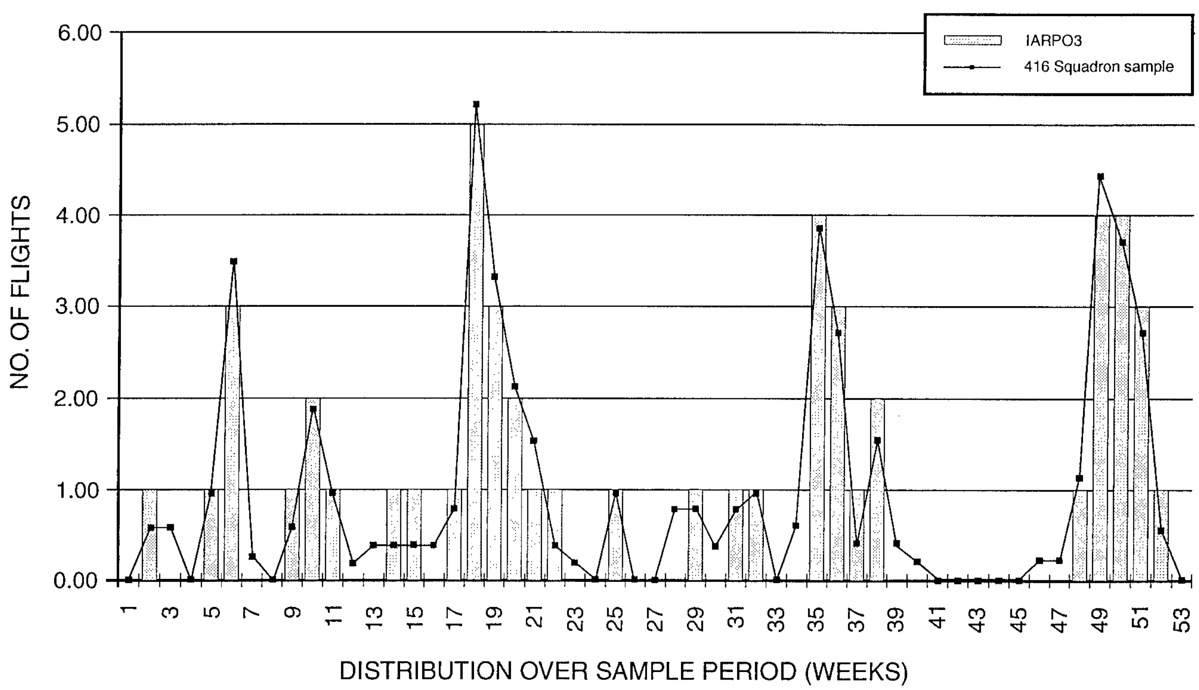


Figure 12. Mission C Sequence

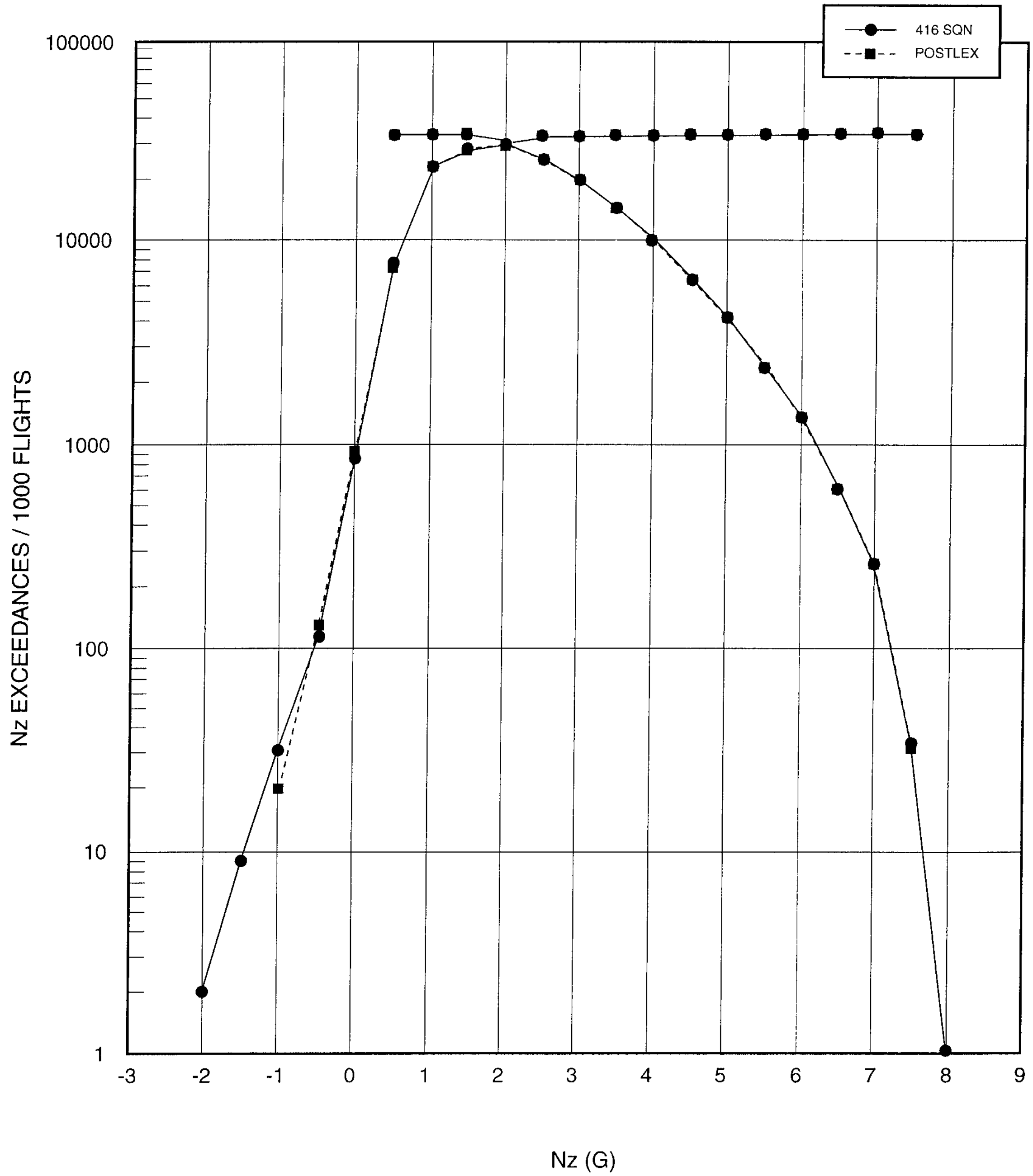


Figure 13. All Missions

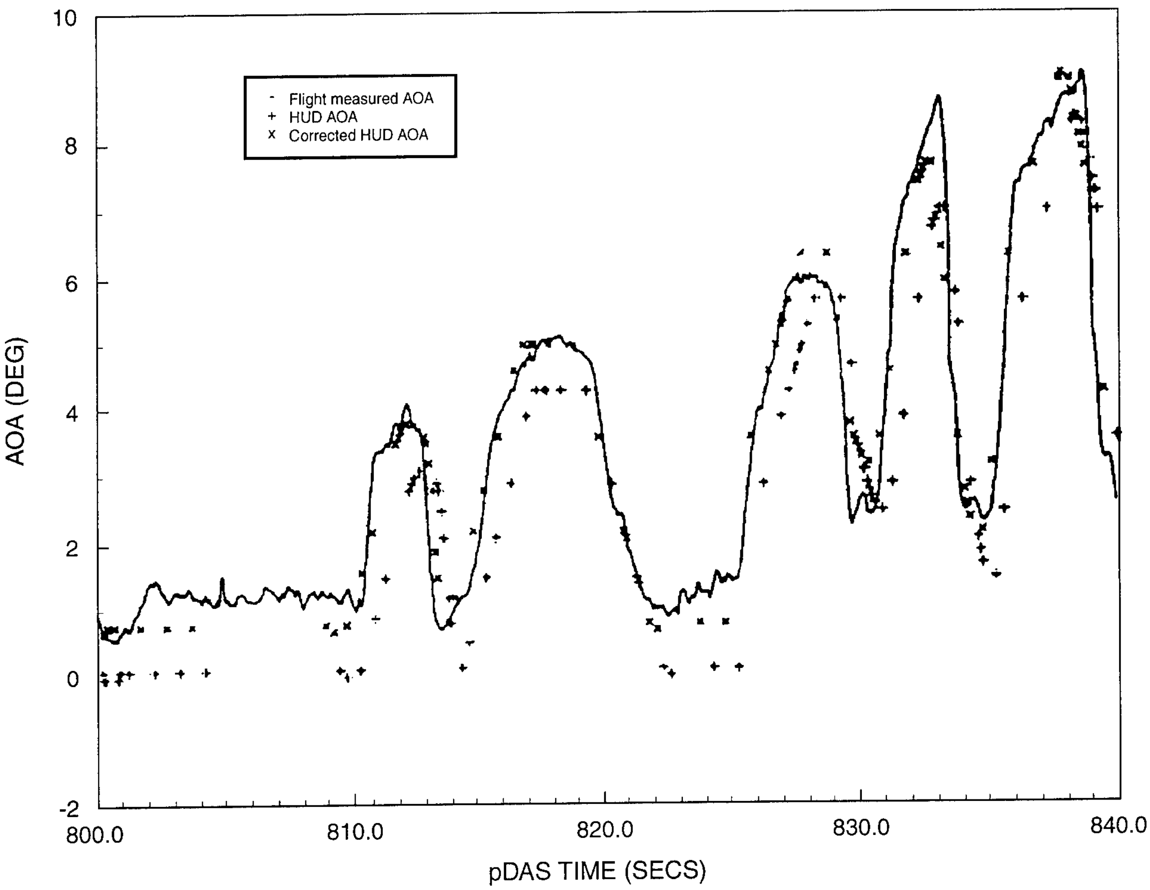


Figure 14. Comparison of Flight Measured AOA and HUD AOA.

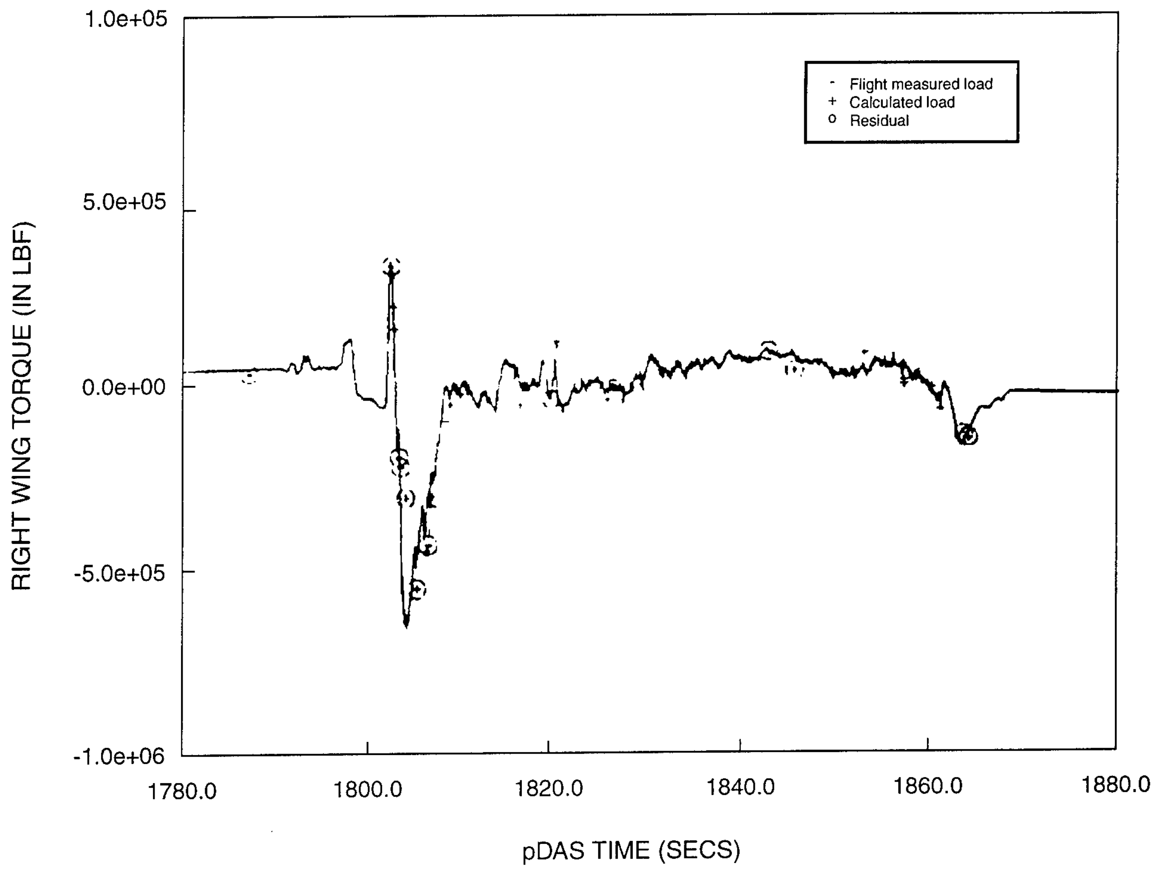


Figure 15. Comparison of Predicted and Flight Measured Right Wing Root Torque.

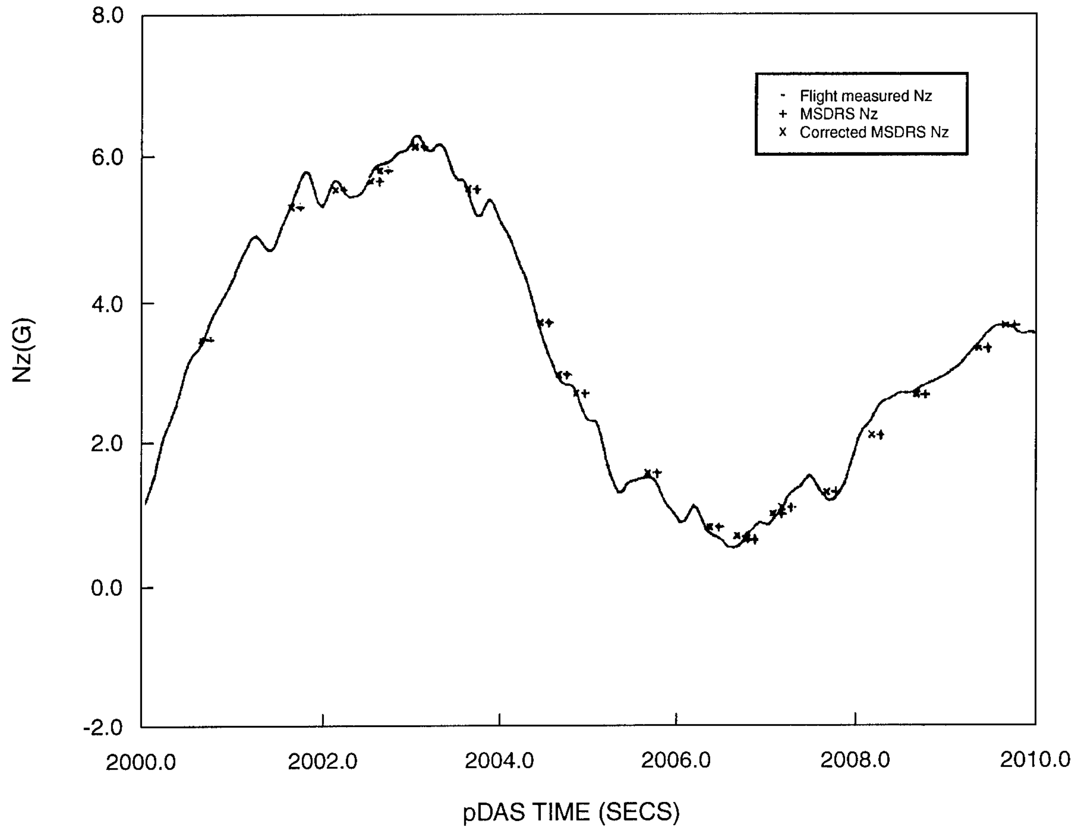


Figure 16. Comparison of Flight Measured and MSDRS Nz

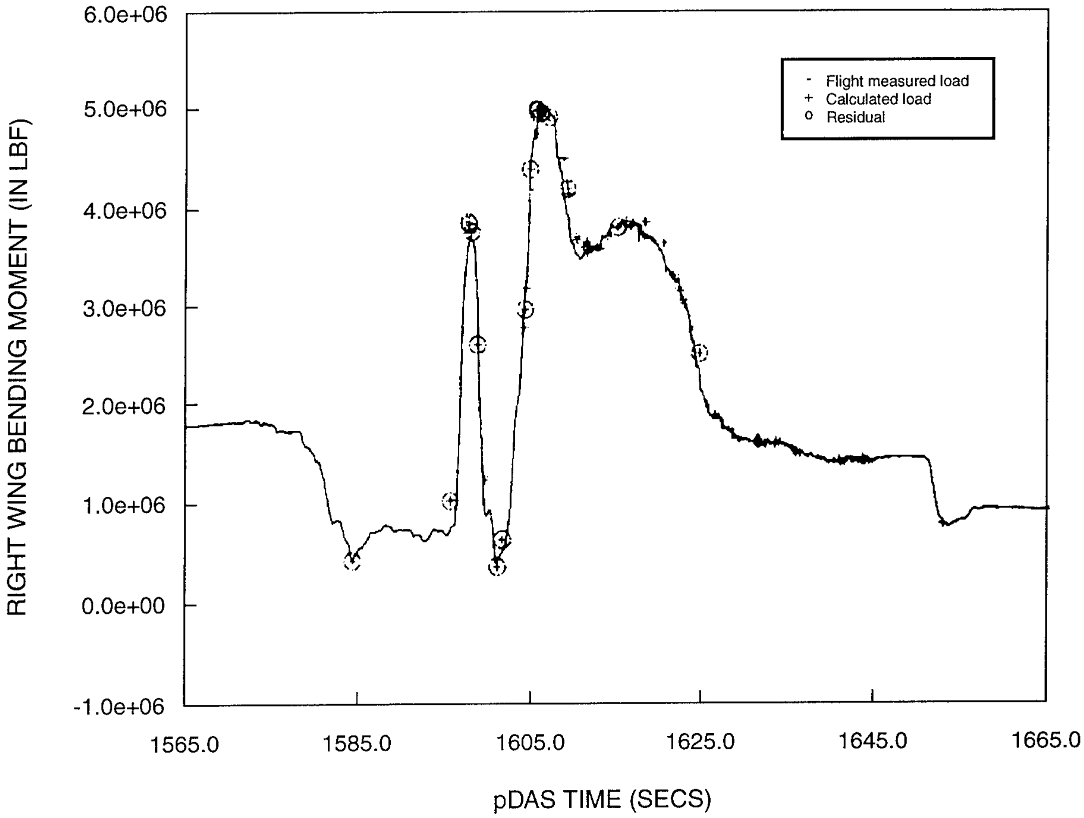


Figure 17. Comparison of Predicted and Flight Measured Right Wing Root Bending Moment.

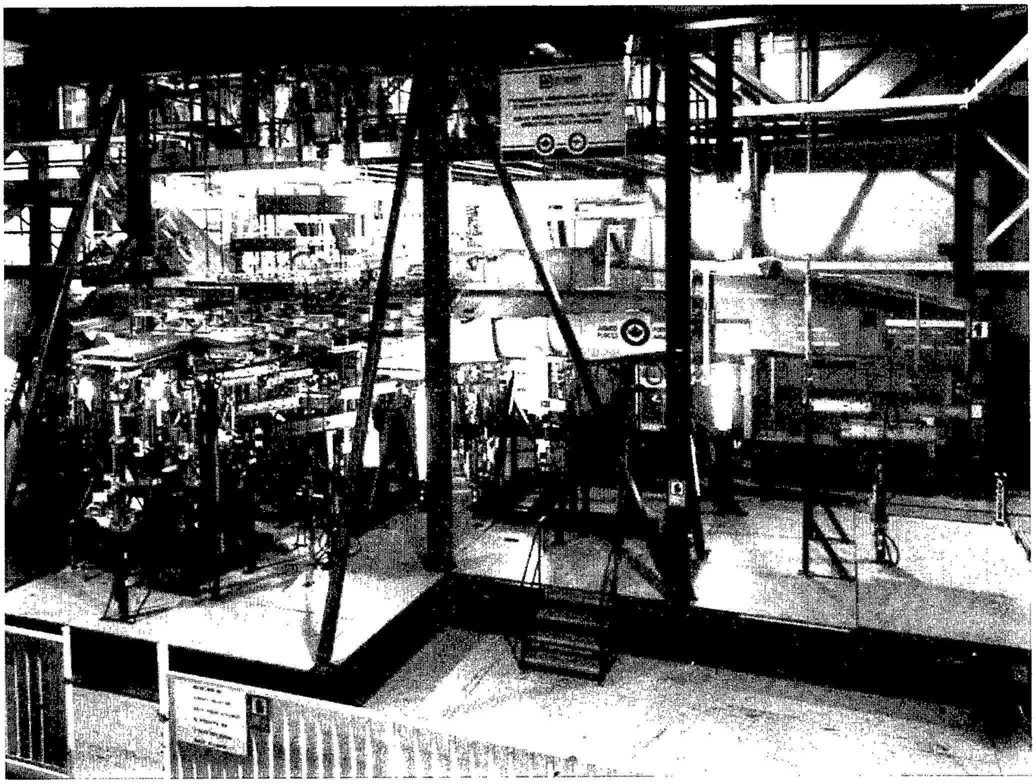


Figure 18. Centre Fuselage Test

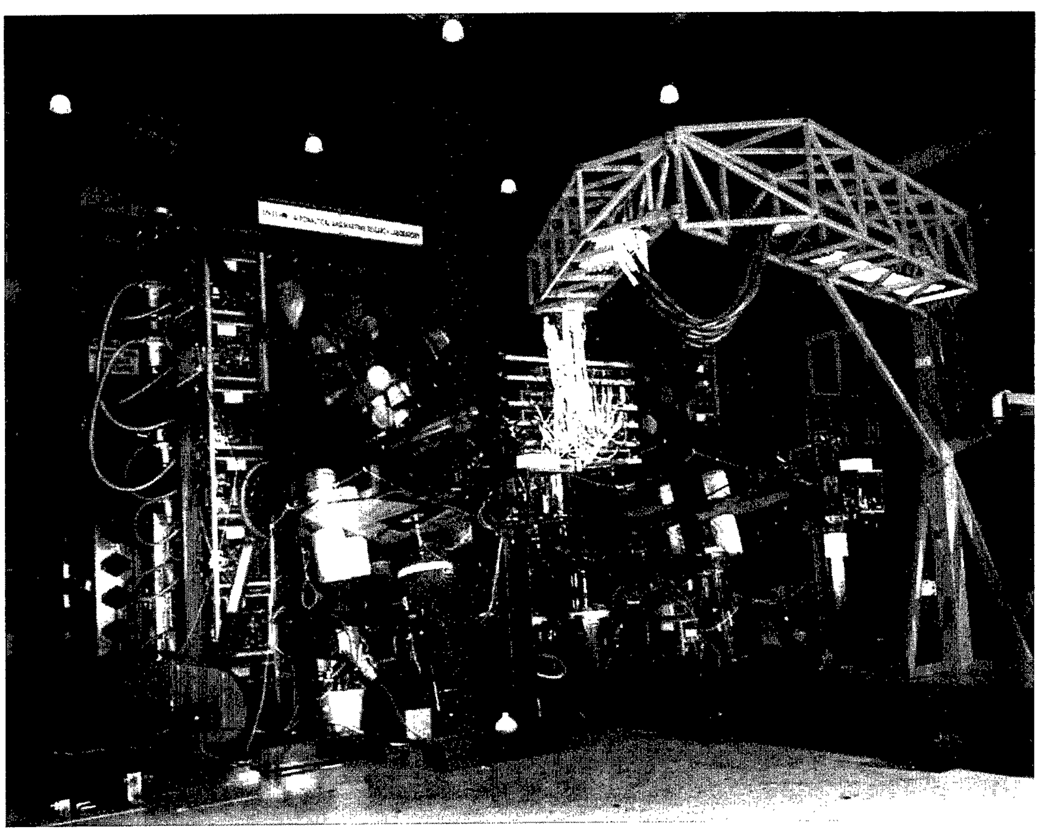


Figure 19. AFT Fuselage Test

F-16 SYSTEM / STRUCTURAL UPGRADES

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SUMMARY

SYSTEMS UPGRADE

Several system upgrades have occurred throughout the life of the F-16 aircraft. This paper briefly discusses various system upgrades including navigational equipment, communication equipment, radar, stability and control, flight control system, and engines.

STRUCTURAL UPGRADES

The F-16 was originally designed to be a lightweight fighter with a service life of 8,000 flight hours. Due to the usage being more severe than design and an increase in weight, several structural modifications were necessary to keep the F-16C/D in service. The structural modification program known as "Falcon Up" is being accomplished by several countries for F-16A/B/C/D aircraft. This program began in June 1993 for USAF and will complete around the year 2001. This paper discusses each of the structural problems and the modifications necessary to reach an 8,000 hour service life.

INTRODUCTION

Development of the F-16 began in the mid 1960s and has continued to evolve. Today, there are approximately 4,000 F-16 aircraft worldwide with production ongoing. With changes in usage, threat, missions, and technologies, several versions of the F-16 have been manufactured and various capabilities have been incorporated in production throughout the various models and "Blocks" (i.e. Blocks 1/5/10/15/20 are A/B aircraft and Blocks 25/30/32/40/42/50/52 are C/D aircraft. A and C models are single-seat aircraft and B and D models are two-seat aircraft). This paper discusses the evolution of systems and structures on the F-16 aircraft.

1.0 NAVIGATIONAL UPGRADES

Navigation equipment is used on F-16 aircraft to provide destination and stabilization signals in support of target acquisition, flight control, ordinance delivery, and typical range, bearing, and time to destination computations. The F-16 navigation system is composed of an Inertial Navigation System (INS), Global Positioning System (GPS), and

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Master Navigation Filtering (MNF) algorithm dependent upon the particular F-16 Block set. Each subsystem and its integration into the F-16 fleet is discussed below.

1.1 Inertial Navigation System (INS)

The INS is a self-contained, all-attitude navigation system providing outputs of linear and angular acceleration, velocity, position, heading, attitude, baroinertial altitude, body angular rates, and time tags. The first inertial navigation system, Singer's SKN-2416, was integrated into the F-16 Block 5 during 1975 and continued into the Block 15 and the first Blocks 25/30/32 aircraft until 1984. It was designed to be a medium accuracy, 0.8 nautical miles/hr, INS which met or performed better than specification. To improve the reliability, the Litton LN-39 INS was selected for integration into Blocks 25/30/32 aircraft in 1984, F-16 Blocks 40/42 aircraft in 1988, and F-16 Blocks 50/52 aircraft in 1991. Litton's LN-39 provided comparable performance with Singer's SKN-2416 but offered approximately three times the reliability (150 hours mean time between failures [MTBF]). In 1992, Honeywell's H-423 and Litton's LN-39 standard Ring Laser Gyro (RLG) INS with F-16 unique software were selected for integration into the F-16 Block 50D. They were to be alternate and interchangeable (form, fit, function) replacements for Litton's LN-39. The RLG technology was selected because its reliability was expected to exceed the LN-39 reliability by an order of magnitude. Still in operation today and retrofitted into F-16 Blocks 25/30/32 aircraft, both Honeywell and Litton RLG INSs have confirmed the expectation of high reliability and have demonstrated an approximate MTBF of 3,500 hours.

1.2 Global Positioning System (GPS)

GPS is a space-based radio-navigation system employing satellites to transmit timing signals to GPS receivers which are enabled to determine position, velocity, and time. GPS was first integrated into the F-16 Block 40/42 aircraft in 1988 and then in F-16 Block 50/52 aircraft in 1991.

The F-16 C model, single-seat Block 40/50 aircraft, is comprised of a Rockwell-Collins GPS Receiver 3A, an E-Systems Antenna Electronics Unit (AE-1), and E-Systems & Ball Aerospace Controlled Reception Pattern Antenna (CRPA). The AE-1 supports the F-16 CRPA and is designed to produce nulls in the antenna array gain pattern in the direction of the jamming signal sources; split a composite RF (Radio Frequency) signal into L1 and L2 signals; down convert signals from RF to IF (Intermediate Frequency); and output L1 and L2 signals on separate coaxial cables. The F-16 conformal CRPA is a seven element antenna with six elements symmetrically arranged around the center element providing anti-jamming capability.

The F-16 D model, dual-seat Block 40/50 aircraft, is comprised of a Rockwell-Collins GPS Receiver 3A, a Rockwell-Collins Antenna Electronics Unit (AE-4), and a Rockwell-Collins Fixed Reception Pattern Antenna (FRPA). The AE-4 supports the F-16 conformal FRPA and is designed to accomplish the same features as the AE-1 except the AE-4 does not contain the anti-jamming feature. The F-16 conformal FRPA is a single element antenna which does not provide anti-jamming protection.

1.3 Master Navigation Filter (MNF)

The Master Navigation Filter, designed by Lockheed Martin Tactical Aircraft Systems, is a Kalman Filtering algorithm hosted in the Generalized Avionics Computer. The MNF was integrated into the F-16 Block 40/42/50/52 aircraft during the same time period as GPS. The MNF integrates navigation data from multiple sensors, including INS and GPS, to derive a system navigation solution consisting of aircraft position, velocity, and heading information.

1.4 Embedded GPS/INS (EGI)

The Honeywell H-764G was selected in 1995 to replace the current RLG INS and provide the F-16 Block 25/30/32 aircraft with GPS capability. The F-16 C and D model configurations will be the same for Blocks 25/30/32 as they are for the Blocks 40/42/50/52 regarding Antenna Electronic Units and GPS antennas. The EGI contains a Honeywell RLG, Rockwell-Collins GPS Embedded Module, and a Honeywell designed Kalman Filter used to blend the INS and GPS sensor data. The EGI is anticipated to be fully integrated and fielded by 1999.

2.0 COMMUNICATION EQUIPMENT UPGRADES

Communication equipment is used on the F-16 aircraft to allow communication between aircraft and ground forces. The F-16 has a communication system which is comprised of data modems and HF/VHF/UHF radios. Each of these is discussed below.

2.1 Data Modems

Late in the 1980s, the USAF defined requirements to automate the process of transmitting the target coordinates and characteristics of threat emitters in prosecution of the Suppression of Enemy Air Defenses (SEAD) missions. The existing US tactical data modem at that time was the Automated Target Hand-off System (ATHS) used by the US Army and Marines. Hand-held data modems had been and were being developed to communicate with the ATHS via existing service link and message protocols and standards. But the ATHS was limited to 1,200 baud and the USAF desired to increase both its data rates and modulation types. This led to a program to upgrade the ATHS capability. The ATHS was dubbed the ATHS I while its increased capability relative would be called ATHS II with its data rates increased to 16K baud and adding digital and secure digital data modulation. Although originally begun as a commercial contract, programmatic, cost, and schedule considerations resulted in the program being taken over by the Naval Research Laboratory (NRL) in 1990. NRL already had a working Modular Airborne Tactical Terminal (MATT) and included the desired increased ATHS II capabilities; the effort was renamed Improved Data Modem (IDM). Meanwhile, the US Army and the Air Force A-10 program office took interest in the IDM, and a joint development began which infused even more capability with expanded software into the IDM effort. Since then, IDM has included in its Army software about 80 message types, while the USAF and Foreign Military Sales (FMS) software includes the SEAD message, 8 Close Air Support messages, and an intraflight data link message/data set for up to 16 aircraft.

IDM began first field tests in 1992 and first production units were received in 1994. The IDM and its software is compatible with the existing USAF hand-held Digital Communications Terminal (DCT) and the Base Communications Terminal (BCT) as well as the US Army's Forward Entry Device (FED), Data Entry Device (DED), and Marine DCT. F-16 Block 50s have over 136 IDMs installed with a planned total of about 185. Additional FMS deliveries have been made. The US Army has received over 300 IDMs building to over 1,200 total. Over 143 FMS IDMs have been delivered with over 500 on order. The US Air National Guard and Air Force Reserve (ANG /AFRES) desired to data link their aircraft as well, but favored the US Army Enhanced Position Location Reporting System (EPLRS) radio/modem set. The EPLRS with the ANG/AFRES modified software called Situation Awareness Data Link (SADL) also provides intraflight data link capability as well as the function of directly linking with Army and Marine equipped EPLRS and PLRS units. The EPLRS is a synchronous Time Division Multiple Access terminal which uses direct sequence pseudo noise bandwidth spreading techniques as well as frequency hopping with intrinsic crypto security (similar to NATO Link-16 but with different frequencies and modulation formats. Several hundreds of ANG/AFRES Block 25/30 F-16s as well as over 200 USAF Block 30 F-16s are to receive the EPLRS/SADL equipment.

2.2 **HF Radios**

Some F-16s, notably the USAF Air Defense Fighter and certain foreign military sales variants, employ HF radios for long haul, beyond-line-of-sight communications. The actual range depends upon atmospheric conditions. Due to the size, power, and weight of HF radios, such is not typically a choice for shorter range tactical communications.

2.2 **VHF Radios**

The ARC-186 has long been a part of the F-16 communications suite and is common across all USAF and most FMS blocks. The F-16 VHF Receiver-Transmitter (RT) is remotely located in the left forward avionics bay. It is operated via the Up Front Controls (UFC) which utilize a Digital Electronic Entry Unit to translate the UFC MIL-STD-1553 data into the form of serial data which the ARC-186 requires for its control and display. There have been two recent initiatives to provide frequency hopping capability via the US Army Single Channel Ground Air Radio System (SINCGARS) into the F-16 VHF communications. The first attempt ended approximately in 1989-1990 when the contractor was unable to fulfill all of the requirements of the specification (ARC-201). A second attempt was made starting in 1991 (ARC-222 or Airborne SINC-GARS). The ARC-222 can fulfill the requirements of the specification; however, in 1995, the USAF determined through its Fighter Configuration Plan process that adding VHF frequency hopping capability to the F-16 was no longer a top priority. The F-16 portion of the program was canceled.

2.3 **UHF Radios**

F-16s beginning delivery in 1978 were equipped with single channel UHF radios since the frequency hopping Have Quick UHF radio capability was not available until the early 1980s (Have Quick I). The desire for faster hop rates and added ease of operation existed, leading to an enhanced Have Quick in the late 1980s. These enhancements included an extended memory board which allowed additional Word-Of-Day (WOD) keys and frequency hop-sets to be loaded and stored. By the mid-1990s, night vision illumination system compatible controls and displays were available and additional electronic key fill of WODs, frequency presets, and hop-sets were available as well. The F-16 ARC-164 UHF radio is panel mounted with integral control and display, and is also controllable via the UFC in a similar manner as the ARC-186 VHF radio.

3.0 **RADAR SYSTEM UPGRADES**

The F-16C/D aircraft employs the AN/APG-68 Fire Control Radar System. The AN/APG-68 Fire Control Radar is an X-band, all-weather, multimode fire control radar featuring extensive air-to-air (A/A) and air-to-ground (A/G) capabilities. It is designed to establish airborne situation awareness and enhance navigation efficiency, as well as provide decisive air or surface target detection and track capability. Air-to-air modes employ sophisticated digital Doppler signal processing techniques to detect and display airborne targets in multi-target tracking for situation awareness and for weapon employment in the presence of sophisticated enemy Electronic Counter Measure techniques. Air-to-ground modes provide mapping, target detection, target tracking, and line-of-sight ranging capabilities to support accurate navigation and weapons delivery.

Beginning in the mid 1970's the first A/B F-16s were introduced into field service. They were equipped with a single transmitter mode, coherent pulse Doppler radar, the Westinghouse, AN/APG-66 system. This system offered 14 flexible, multirange A/A and A/G pilot selectable modes. The hardware consisted of six line replaceable units (LRUs) with a ultraviolet programmable read only memory loadable software operational flight program (OFP). By 1978, a much improved C/D F-16 aircraft version was introduced to the field. The new aircraft was fitted with a dual mode transmitter equipped, coherent pulse Doppler radar, the Westinghouse, AN/APG-68 system. This system also offered 14 flexible, multirange A/A and A/G pilot selectable modes with a reduced 5 LRU hardware set. Both USAF and FMS versions were created. The USAF version included a dual mode, medium and low pulse repetition frequency (PRF) transmitter with higher average transmit power settings. This added capability was used in support of the velocity search and velocity search with ranging modes in support of the Pulse Doppler Illumination (PDIL) missile system and in special Single Target Track (STT) submode functions. The APG-68 radar also offered a greater frequency agility capability than the older APG-66 radar. The older APG-68 models did use Block Oriented Random Access Memory (BORAM), N-MOS based technology that still proves a maintenance problem when reloading the software OFP. Various 3 and 1 card Electronically Erasable Programmable Read Only Memory (EEPROM) shop replaceable unit (SRU) board changes were retrofit to some units to increase non-volatile memory space from 416K words to 512K words and to alleviate some reload problems. All Block 40 F-16 aircraft have the new 512K EEPROM boards, but the 25/30 Block A/C have a mixture. The last OFP update was made on the older 25/30/40 Block jets via SCU II and SCU III (OFP designations) along with other avionics OFP updates. A 3 Digit Radar program (3DR) was also introduced in 1994 to upgrade the radar reliability to achieve an MTBF of 100 + hours, a first for a fire control radar. The program to monitor maintenance activities is still in place and shows a screened response of 242 hours system level MTBF. As with the memory change, not all Block 25/30/40 aircraft radar received all the engineering changes to affect this performance. Changes have been inserted on an attrition basis only for cost efficiency. From 1992 to 1993, the F-16 SPO developed an Advanced Programmable Signal Processor (APSP). This form/fit /function replacement for the older processor went into the new Block 50 aircraft, AN/APG-68 radar. It provided greater reliability (1,000 hour MTBF), cost about 25 percent less to make, used Very High Speed Integration Circuit (VHSIC) technology, had 14 SRUs rather than 38, weighed 60 percent less and required half the power and cooling, doubled the throughput, and used non-volatile memory storage and Static Radom Access Memory (SRAM) storage. It made possible the introduction of the new velocity search with ranging mode, better Electronic Counter Counter Measure (ECCM), a more robust dual target situation awareness mode (SAM), and provisions for a future data editor.

The current software version 8030 has been extensively tested and found to be the most robust airborne fire control radar in the US inventory. A recent 8031 version improves dual target SAM capability. Further OFP work is underway for version 8040 in 1996 and version 8050 in 1997, with such new modes as a mutual interference data editor.

For the APG-66 radar, Westinghouse was contracted to redesign parts of this radar for the Mid Life Update (MLU), European Participating Government (EPG) program. To increase reliability, improvements were made to the antenna, the transmitter, radar computer (RC), and array processor (AP) LRUs. Utilizing VHSIC technology similar to that used in the APSP and a new processor/software architecture, they combined the RC and AP into one LRU processor and achieved a very flexible radar set, much improved over the older APG-66 set. This system provides an expanded radar mode set, improved performance against mutual interference, software tool flexibility, and inherent trouble-shooting capability.

4.0 STABILITY AND FLIGHT CONTROL UPGRADES

The F-16 combines advanced aerodynamic features and a fly-by-wire flight control system to produce high maneuverability. The flight control system provides stability augmentation which permits the use of relaxed static stability in order to realize the performance benefits of reduced trim drag. The flight control system also uses several limiters to allow the pilot to maneuver the aircraft to its full capacity while preventing departures from controlled flight and structural overstress. A Stores Loading Category switch in the cockpit allows the pilot to select the limiters appropriate for the external stores configuration the aircraft is carrying.

4.1 Increased Area Horizontal Tail (IAHT)

During the early years of the F-16 program, concerns arose about the future price and availability of the titanium used to make the horizontal tails. Studies were initiated to develop a revised design. In addition to developing new structure, these studies concluded that the tail area should be increased 30 percent. This increased area was expected to result in better takeoff performance, increased resistance to departure from controlled flight, relaxed aft CG limits, and an increase in maneuvering capability with various external stores loadings. Subsequent flight testing verified these benefits. Flying qualities at cruise conditions with the IAHT were at least as good as with the original horizontal tail. Departure resistance was substantially improved. As a result, flight limits for many stores loadings were upgraded to CAT I (full air-to-air limits). This included all loadings formerly designated as CAT II, a restricted set of air-to-air limits which required the pilot to monitor angle-of-attack (AOA) while maneuvering. However, the IAHT aggravated some flying qualities deficiencies which were already known to exist in the power approach configuration. This included the tendency of the aircraft to over-rotate and strike the tail on the runway during aero-braking. As a result, new power approach control laws were developed and implemented which corrected these deficiencies. The IAHT was incorporated in Block 15 and later production aircraft. It has also been retrofit to many earlier models. Incorporation of the IAHT significantly reduced the aircraft loss rate in out-of control mishaps.

4.2 New Yaw Rate Limiter

Beginning with Block 30D, an enlarged engine inlet was incorporated in F-16 aircraft equipped with the General Electric F110 engine. Based on wind tunnel testing and analysis, this new inlet was not expected to significantly degrade aircraft handling qualities. However, high AOA testing showed that the inlet degraded both departure resistance and deep stall recovery characteristics to an extent that made improvement mandatory. A recalibrated AOA limiter effectively countered the reduction in departure resistance. However, the principal USAF using command was concerned that this feature would also reduce instantaneous turn rate performance in aerial combat. Additional flight test showed that departure resistance was increased at lower altitudes. Accordingly, the recalibrated AOA limiter was not incorporated in Block 30 aircraft and flight limits were based on test results at lower altitudes. The recalibrated limiter was incorporated in Block 40 and later aircraft to offset expected additional degradation in departure resistance due to increased weight.

Additional flight control law changes were required to remedy the degradation in deep stall recovery caused by the large inlet. Simulation studies showed that changes to both the time constants and gains used by the flight control system's yaw rate limiter function would be effective in restoring recovery characteristics. This analysis was verified by flight test. Additional flight testing showed that the new yaw rate limiter was also beneficial for F-16 aircraft with the small engine inlet, and also for aircraft with the original horizontal tail. As a result, the modified yaw rate limiter was both incorporated in production and retrofitted to all existing F-16 aircraft.

4.3 Improved Digital Flight Control System (DFLCS) Control Laws

A digital flight control computer (DFLCC) was incorporated in the F-16 beginning with Block 40. The main reason for its incorporation was to accommodate an Automatic Terrain Following (Auto TF) function. However, it was also expected to make flight control law changes quicker and cheaper since only software changes would be involved.

Over the next several years, a number of control law improvements were developed. The major improvements included a revised autopilot to improve roll-to-wings level performance in Auto TF with asymmetric stores; an increase in stability margins to reduce a tendency for aeroservoelastic limit cycle oscillations; and handling qualities improvements. The handling qualities improvements included features intended to further enhance departure resistance and deep stall recovery and miscellaneous small changes. Because of funding limitations, all of these changes were bundled into one flight control software release. As a result, testing and development stretched out over a long period of time as small problems were found and corrected. As a result, the final flight control laws were developed too late to be incorporated in production USAF aircraft. However, incorporation is proceeding in remaining FMS production, and the new control laws will be retrofit to all other DFLCS equipped aircraft.

5.0 ENGINES (PRATT & WHITNEY)

Four engines from the F100 engine family power many of today's F-16 aircraft. The F100-PW-200 engine powers most pre-Block 30 F-16 aircraft with a maximum thrust of approximately 24K lbs. The F100-PW-220 engine retains a similar maximum thrust capability as the PW-200 engine but has several reliability and maintainability improvements in controls, accessories, and hardware. The PW-220 engine is installed on Block 32 and 42 F-16 aircraft. To provide the same improvements to the older PW-200 engines, many have been upgraded to the F100-PW-220E configuration. The PW-220E engine primarily powers pre-Block 30 aircraft but can also be used on Block 32 and 42 aircraft. The F100-PW-229 engine provides a significant improvement in performance with a maximum thrust of approximately 29K lbs. PW-229 engines power the newer Block 52 F-16 aircraft.

5.1 Low Pressure Turbine (LPT) Issues

The F100 engine family has exhibited several low pressure turbine issues which have impacted F-16 operational usage over the last several years. In 1993, the F100-PW-229 experienced two fourth stage turbine blade fracture incidents on F-15E aircraft. Four fractures of a earlier configuration fourth blade occurred during development testing. This history indicated that the blade lacked sufficient margin to tolerate some operational conditions. Analysis by P&W determined that one of the failures was due to a material anomaly in a blade. An inspection was then developed by P&W and implemented in the field which allowed USAF personnel to inspect for similar material defects. In addition, the allowable defect size was reduced and inspections enhanced in production to prevent another occurrence. Preliminary analysis of the other failure by P&W indicated high dynamic pressure in conjunction with high cycle fatigue (HCF) were contributing factors. As such, a flight restriction of 550 knots and a maximum usage of 400 Total Accumulated Cycles for the LPT were imposed on PW-229 powered F-16 aircraft. To further reduce short-term risk and eliminate the flight/usage restriction, an interim blade was designed and retrofit into F-16 PW-229 engines. Ultimately, a robust fourth blade was designed to reduce steady stress and vibratory responsiveness. More recently, the F100-PW-220/220E engine family has experienced 10 LPT incidents since May 1995. Most of the failures have been attributed to creep of the third stage turbine blade. Creep refers to the tendancy of rotating turbine engine components to grow during operation, and is a function of operating temperature, steady state stress, and material capabilities. As the third stage blades creep, the tip shrouds curl, inducing stress in the blade tip. Ultimately, cracking occurs in the tip and propagates to failure. To mitigate short-term risk, several inspections have been conducted on operational engines with recurring inspections introduced until redesigned hardware can be fielded. Another failure mode that has been identified is a HCF/LCF interaction which can lead to fracture of the fourth stage turbine blade. Steady stress combining with a vibratory stress (e.g., augmentor instability) causes eventual fracture of the blade. As with the third blade, recurring inspections have been implemented to reduce risk until redesigned fourth blade and disk hardware can be fielded.

6.0 ENGINES (GENERAL ELECTRIC)

Two engine models from the F110 family power the more recent F-16 aircraft. The F110-GE-100 engine powers Block 30 and 40 F-16 aircraft with a maximum thrust of approximately 28K lbs (Sea Level Static). The "Increased Performance Engine," F110-GE-129 powers the newer Block 50 aircraft with a maximum thrust of 28.7K lbs (Sea Level Static). The -129 engine provides up to a 30 percent thrust increase, depending on flight condition, over the older -100 engine. The thrust increase comes from higher rpms and combustor exit temperatures. The -129 engine also has a digital electronic control while the -100 engine has an analog control which is facing obsolescence. A major effort is starting to replace all -100 analog controls with digital electronic controls.

6.1 Airseal Failures

Several HCF issues have beset the F110 family over the past three years and have caused engine failures. The most difficult problem from a technical standpoint were engine failures caused by failures of the "threetooth airseal" in the turbine section. These failures were originally attributed incorrectly to thermal instability. An intensive hardware investigation, analysis and test program revealed that the failures were due to a complex eccentric rub induced high cycle fatigue cracking mode encountered during the initial break-in runs. This failure mode was the result of opening up the clearances between the seal teeth and the mating honeycomb in an effort to address the thermal instability concern. All low time engines were grounded for several months and production lines were also stalled. A fix involving a return to the original seal clearances and the addition of a "sleeve" damper finally resolved the problem.

6.2 1st Stage Fan Blade

HCF cracking of the -129 first stage fan blade dovetail and extreme sensitivity to foreign object damage (FOD) has been another recent problem. Another intensive investigation effort revealed that this problem was due to a combination of factors; dovetails out of tolerance due to a manufacturing problem, FOD, inlet guide vane/flap offset (blades get excited by flow disturbances), and midspan shroud damping variations. For risk management, the blades are currently being replaced every 700 hours and have severely tightened FOD inspection/ serviceability limits which are very burdensome to the field. Manufacturing changes have also been introduced to ensure a better blade. A fix has been developed involving the addition of an underplatform damper, slight change to the midspan shroud configuration, and cutting back the blade leading edge.

6.3 2nd Stage Fan Blade

HCF cracking of the second stage fan blade and disks is a current problem in the -100 engine. Currently, a 200-hour phase inspection has been instituted with specially developed ultra-sonic probes. As data indicates a slow crack propagation rate, this inspection adequately manages risk. Investigation indicates that vibratory stresses cause dovetail wear/fretting which in turn degrades fatigue properties and increases stress. The degraded fatigue properties coupled with increased stress leads to HCF cracking. A fix has not been developed but is currently being evaluated to include blade dovetail and disk rework to redistribute stresses.

7.0 USAF STRUCTURAL UPGRADES

The F-16A/B Block 10/15 aircraft was originally designed and certified to an 8,000 hour service life requirement based on a usage environment developed from historical data. Changes in mission mix (Figures 1&2) and an increase in usage severity (Figure 3) and weight, led to a decrease in service life. With the severity of the A/B usage not fully understood, and budget constraints for structural upgrades, only minor structural improvements were implemented in going from Block 10/15 to Block 25/30. When these shortfalls were recognized, the service life requirement was relaxed to a goal for the F-16C/D Block 25/30 aircraft. Prior to the production of Block 40 aircraft, actual usage data had been evaluated and the F-16C/D Block 40 requirements and design usage became fully defined reestablishing the 8,000 hour service life requirement. This led to the incorporation of major structural improvements for Block 40 aircraft. During Block 40 production, the Block 30 durability test was still in the first lifetime of test and the Block 40 aft fuselage durability test had just begun. Several more deficiencies were discovered from both full-scale and component tests which led to production incorporated structural improvements for the F-16C/D Block 50 aircraft. The structural improvement program which is comprised of seven Engineering Change Proposals (ECPs) and will retrofit Block 25/30/40/50 is known as Falcon Up (Figure 4).

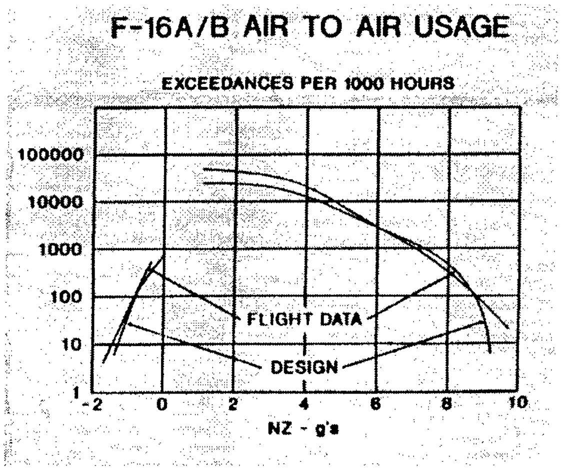

Figure 1

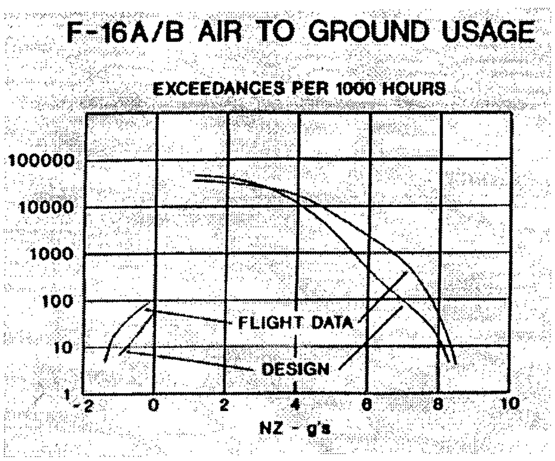

Figure 2

7.1 ECP 1910

In 1989, the F-16C Block 30 full-scale durability test shut down due to a failure in the center fuselage at 7,330 flight hours. The failure occurred at FS 341 (Fuselage Station; measured in inches from nose of aircraft) in the upper to lower bulkhead attachment area known as the fuel shelf joint (FSJ, Figure 5). The center fuselage was replaced and the testing continued. The failed center fuselage section underwent a teardown inspection which revealed several locations of cracking.

The most significant cracks were discovered in the fuel shelf joints, fuel shelf webs, flanges and shear webs throughout the wing carry through upper and lower bulkheads. Other significant areas that were found cracked during this inspection were the main landing gear bulkhead at FS 341 (second bulkhead, first replaced at 3,988 flight hours), upper fuselage skins, and center/forward fuselage longerons.

Approximately one year later, there were indications that the Navy's F-16C Block 30 aircraft were experiencing cracking in the FSJ. A team of USAF engineers and NDI specialists traveled to Miramar to confirm the indications. Cracking was confirmed in the fuel shelf webs and bulkhead flanges in the FSJ area in almost every aircraft that was inspected. This discovery led to the inspection of 12 high time USAF F-16A/C Block 10/25/32 aircraft. The inspection results were similar to those of the Navy.

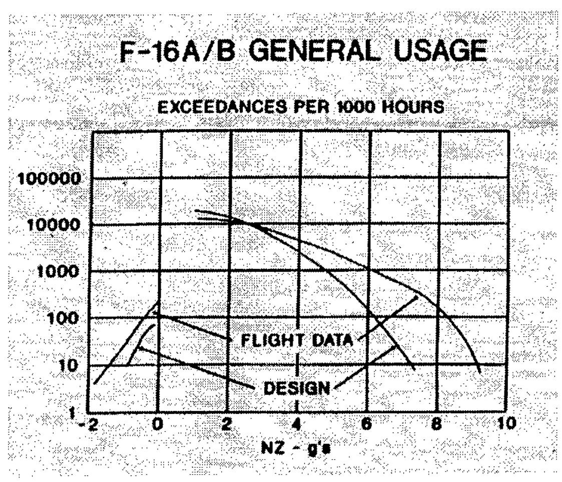

Figure 3

The results of these inspections led to the development of ECP 1910 which would become the backbone of the Falcon Up program.

A component test specimen was designed to represent the FSJ upper to lower bulkhead attachment (Figure 6). The primary load which is most damaging to this area is wing root bending caused by symmetric and nonsymmetric maneuvers. Several com-

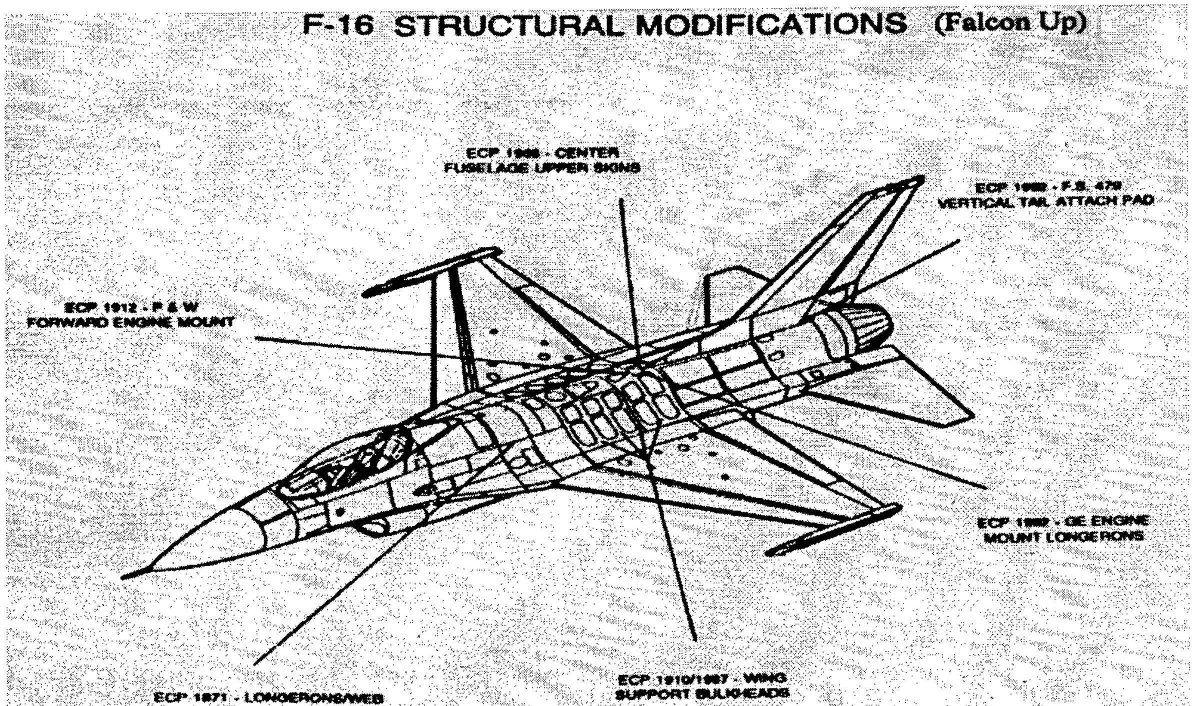

Fig 4

ponent tests were run to achieve a good correlation with the fractographic data that was obtained from the Block 30 full-scale durability test. This crack growth data was used as a baseline to measure the benefits of enhancement options. Once this was achieved, several enhancements were studied to obtain a goal of an additional 5,500 hours of service life. This goal was chosen based upon the estimated aircraft mod time of 2,500 flight hours (e.g. 2,500+5,500=8,000). Several component tests were run varying the configuration to evaluate the effect of crack growth for each enhancement and all enhancements combined (Figure 7). The final configuration of the FSJ for the Block 25/30 aircraft consisted of cold worked holes, necked down bolts, and radius blocks. Cold working the FSJ bolt holes creates a residual compression field around the bolt hole which slightly delays the time to crack initiation and significantly decelerates crack growth through the residual compression zone. The necked down bolt is used to eliminate bearing stress on the edges of the hole which is the second most critical stress component in this area next to bending. The radius block (Figure 8) is a steel block with feet used to distribute the load away from the hole. FS 309 was the only wing carry through bulkhead that did not require radius blocks. Other modifications accomplished under ECP 1910 for pre-Block 40 aircraft were cold working approximately 200 center fuselage bulkhead web penetration holes, doublers bonded and mechanically fastened to 13 shear webs throughout the center fuselage, and replacement of the main landing gear lower bulkhead at FS 341. The FS 341 lower bulkhead was replaced on the Block 30 durability test at 3,988 flight hours and again at 7,330 flight hours. Cracking was discovered in panels A and C (Figure 9) originating out of electrical and hydraulic penetration holes and satellite rivet holes, respectively. The replacement bulkhead has thicker webs, several cold worked penetrations, and fewer satellite holes in panel C.

The thickness of the upper flanges of the wing carry-through bulkheads was increased for Block 40/50 aircraft prior to production. However, several component tests were conducted and the results conveyed a need for cold working. Therefore, the only retrofit modification for Block 40 under ECP 1910 was to cold work the FSJ area at all wing carry through bulkheads.

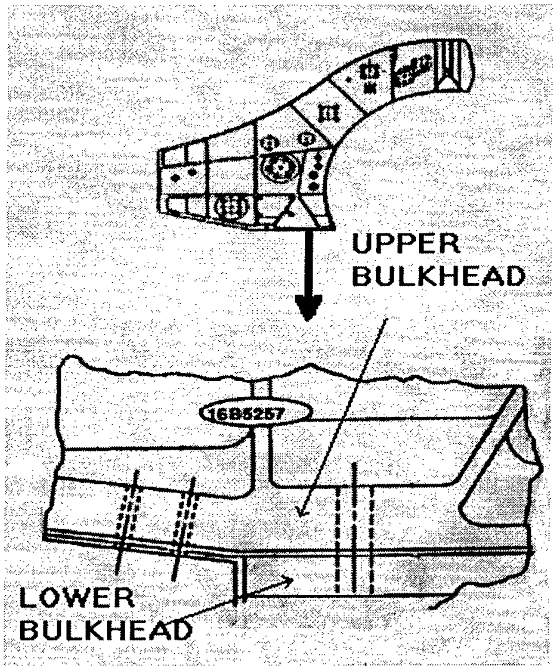

Figure 5

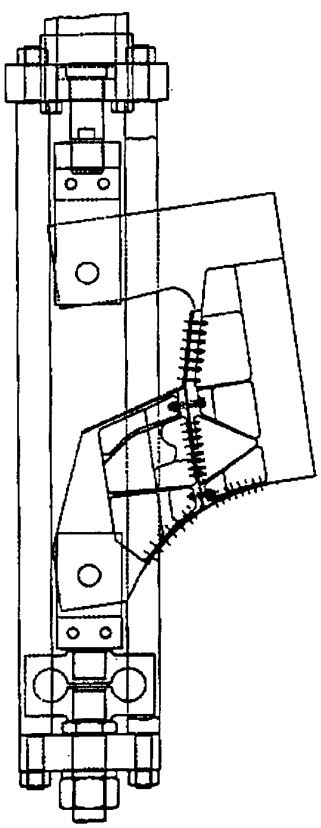

Figure 6

7.2 **ECP 1987**

During the Block 40 FSJ component testing, a damage tolerance deficiency was discovered in the inclined stiffener located in the upper bulkhead at FS 341 adjacent to the FSJ. This area has a long durability life but is very sensitive to initial flaw sizes. The crack initiated out of the .12 inch radius in the inclined stiffener next to a boss which supports a fuel transfer pump. Under ECP 1987, the radius is increased to .5 inches and the boss is machined off. A machined doubler (with an integral boss) and a fitting are bonded and mechanically fastened to the forward side of the bulkhead webs. A steel strap is also fastened to the inclined stiffener in the area where the radius was increased. This modification is accomplished on Block 10/15/25/30/40 and early Block 50/52 aircraft. Later Block 50/52 aircraft were corrected via a production change. The other modification accomplished under ECP 1987 is the cold working of approximately 175 fastener holes located on the lower strake flange of the wing carry through bulkheads. This area was found cracked on the Block 30 full-scale durability test and on some fielded aircraft. Only pre-Block 40 aircraft are affected by this modification due to thickness changes being broken into production at Block 40.

7.3 **ECP 1871**

A center fuselage longeron known as the Thunderbird longeron has been a problem for pre-Block 50 aircraft (Figure 10). Several field failures and test failures had occurred throughout the mid to late 1980's. The redesign process began with the first field failure of a center fuselage longeron which occurred on a F-16A Thunderbird aircraft. The Block 30 full scale durability test experienced the first RHS longeron failure at 2968 flight hours. The failed longeron was replaced with a redesigned longeron which incorporated tapered flanges and an increase

SPECIMEN	CONFIGURATION	FAILURE TIME
*16FTB7006-1A	1/16 O.S.+RB+NDB+CX	15075
*16FTB7006-1B	1/16 O.S.+RB+NDB+CX	8415
*16FTB7006-1B2	1/16 O.S.+RB+NDB+CX	13425
*16FTB7006-1C	1/16 O.S.+RB+S.B.+CX	10955
*16FTB7006-1D	1/16 O.S.+RB+S.B.+CX	9505
*16FTB7006-1E	1/8 O.S.+RB+S.B.+CX	12075
*16FTB7006-1E2	1/8 O.S.+RB+S.B.+CX	13015
*16FTB7006-1F	1/8 O.S.+RB+S.B.+CX	7575
*16FTB7006-1G	1/8 O.S.+RB+NDB+CX	12660
*16FTB7006-1G2	1/8 O.S.+RB+NDB+CX	11000
*16FTB7006-1G3	1/8 O.S.+RB+NDB+CX	8510
16FTB7006-3A	PRODUCTION CX ONLY	9925
16FTB7006-1A	BASELINE NO CX OR SEAL BOND	5000
16FTB7006-3A	BASELINE NO CX OR SEAL BOND	5225
16FTB7006-3B	PRODUCTION CX ONLY	7575
16FTB7006-5	SEAL BOND ONLY NO CX	6500
16FTB7006-1B	BASELINE W/RB NO CX	10500
16FTB7006-3C	PRODUCTION CX W/ RB	13000

* Components were tested to 2,000 flight hours prior to retrofit

Figure 7

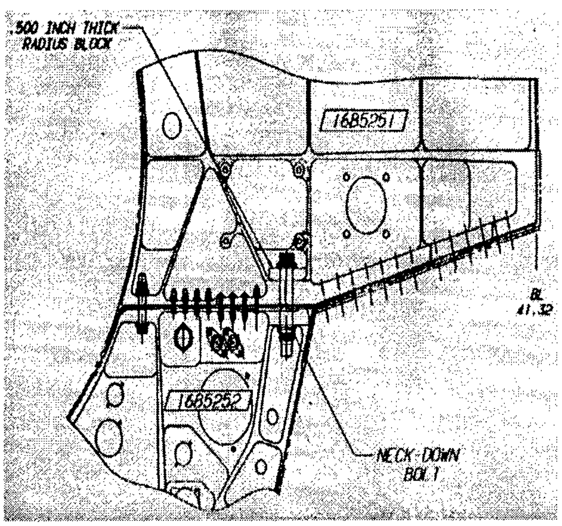

Figure 8

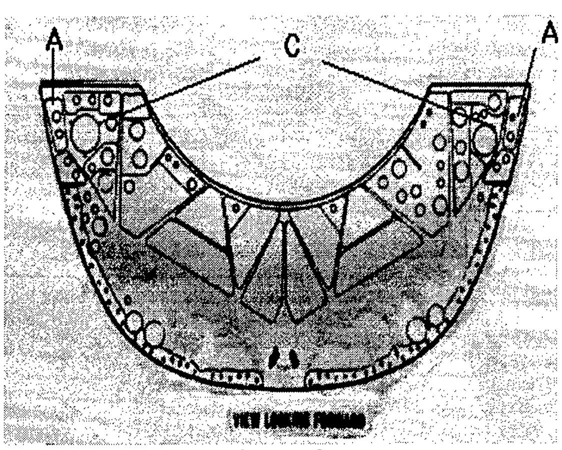

Figure 9

in thickness. The LHS longeron failed at 3988 flight hours and was replaced with the redesigned longeron. Longeron failures continued throughout the remainder of the Block 30 full scale durability test. The longeron design incorporated into ECP 1871, consisted of a new planform, increased thickness, and a material change from an 2024-T62 extrusion to 7475-T7351 machined. ECP 1871 also redesigned the forward fuselage longeron which is located just ahead of the center fuselage longeron. This redesign was required to accommodate more load being drawn into the forward longeron due to the increased stiffness of the redesigned center fuselage longeron. This modification of the forward/center fuselage longerons is applicable to all F-16 pre-Block 50 aircraft and is being incorporated on all USAF Block 40/42 aircraft. ECP 1871 also includes a redesigned center fuselage skin. Two cracks located in the outboard access panel cutouts were found in the Block 30 full scale durability test at 7,330 flight hours. The redesigned skin incorporated new access panel cut out shapes and a tab where the center fuselage longeron mates up. This modification is also applicable to all F-16 pre-Block 50 aircraft and is being incorporated on all USAF Block 40/42 aircraft.

7.4 ECP 1966

The upper center fuselage skins (Figure 11) experienced cracking on the Block 30 full-scale durability test and on the Block 40 aft fuselage durability test. The aft skin was found cracked at 7,330 flight hours on the Block 30 full scale durability test and at 2,317 flight hours on the Block 40 aft fuselage test. The difference in flight hours results from the fuselage bending loads being higher for the Block 40 aircraft. The center skins experienced cracking at 7,330 and 7,558 flight hours for Block 30 test and Block 40 test respectively. There have been several field reports of cracking in the upper aft fuselage skin including three complete failures resulting in significant fuel leaks. Modifications were designed for both aft and center skins which consisted of external doublers, cold working, and internal gussets and longerons (Figure 12). This modification was installed on a Block 40 aircraft and a strain survey was conducted to validate finite element model predictions. This modification is accomplished under ECP 1966 and is applicable to all F-16 pre-Block 50 aircraft and is being incorporated on all Block 40/42 aircraft. Block 50 incorporated an improved design in production.

7.5 **ECP 1962**

The General Electric engine mount longeron (Figure 13) was discovered cracked at 10,000 flight hours on the Block 40 aft fuselage test. This longeron had sufficient durability life but did not meet the damage tolerance requirements. The crack driven by fuselage bending, initiated out of a "race track" shaped

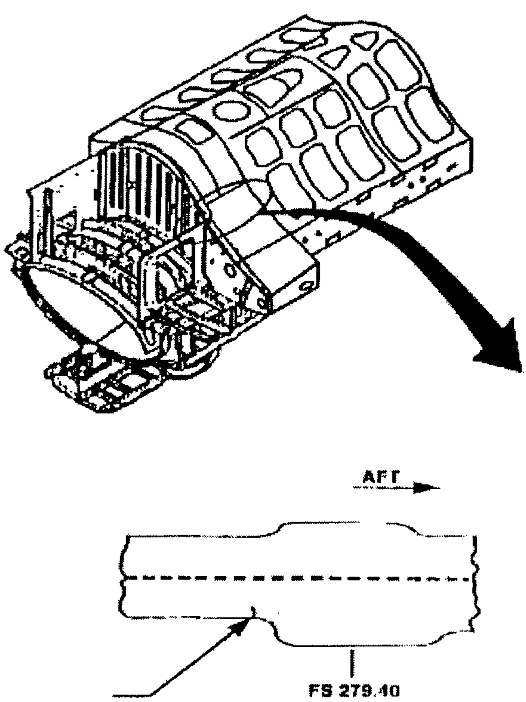

Figure 10

hole. The "race track" hole is cold worked along with three other holes in this area under ECP 1962. Several component tests were conducted to correlate with the test cracking and to evaluate the effects of cold working a non-round hole. This modification is applicable to and being incorporated on all USAF Block 30/32/40/42 aircraft. A new engine mount longeron was designed for Block 50 which included thicker webs and less penetrations.

7.6 **ECP 1912**

The Pratt & Whitney engine mount back up fitting (Figure 14) experienced cracking during Block 40 component testing. Three tests were conducted and two of the components initiated cracks between 4,000 and 5,000 flight hours. The back up fitting is a fracture critical part and does not meet damage tolerance requirements. ECP 1912 changed the material of the fitting from 2124-T851 to 6AL-4V Beta Annealed Titanium. This material change allowed the fitting to tolerate the larger damage tolerance initial flaw sizes with no changes in geometry. This modification is applicable to all pre-Block 50 aircraft with Pratt & Whitney engines and is being incorporated on all USAF Block 42 aircraft.

7.7 **ECP 1992**

In 1989, a USAF F-16A returned from flight with the tip of the vertical tail missing. Inspections later discovered that the FS 479 vertical tail attach bulkhead (Figure 15) had a large crack which initiated at the attach pad radius and propagated through the web to the lower flange. It was later proven that the tail failure was not associated with the bulkhead cracking; tail failure was attributed to jet wake. This incident was the beginning of a fleet wide inspection which would later reveal 90 percent of all USAF pre-Block 40 aircraft had experienced cracking to some degree in the vertical tail attach pad radius. Components were designed and tested to achieve a baseline which developed cracksizes and initiation times which appeared to correlate reasonably well to field experience.

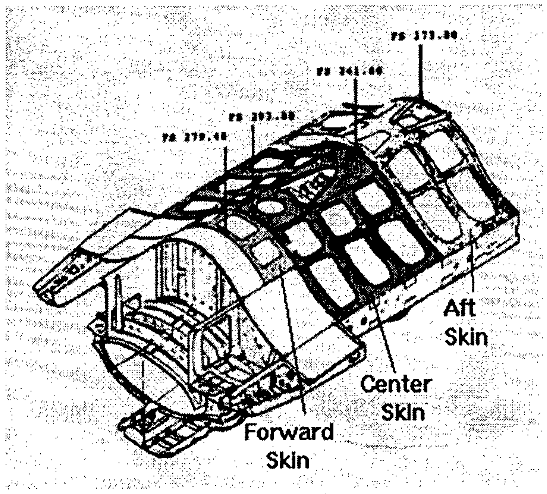

Figure 11

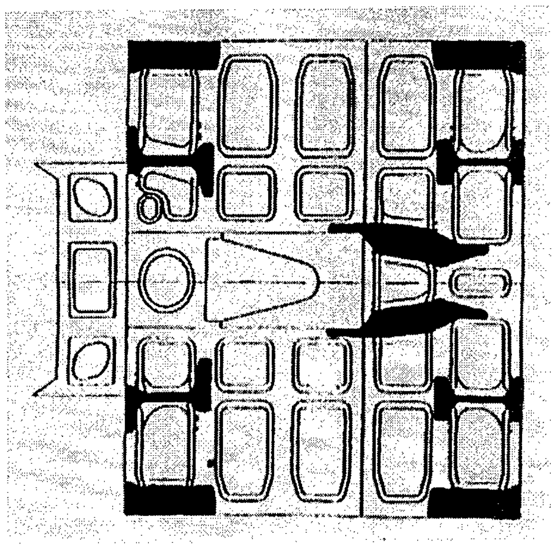

Figure 12

Once a baseline was established, two options were explored: Increase radius from 0.125 to 0.250 inch and increase radius with a maximum undercut of 0.105 inches. The undercut was necessary to remove any remaining crack after the radius was incorporated. Analyses and test results using the flight recorder developed Block 32 usage spectrum concluded that by increasing the attach pad radius from 0.125 to 0.250 inches, the desired service life could be achieved. This modification was accomplished under ECP 1992 and would affect all F-16 pre-Block 50 aircraft with 479 bulkheads which contained 0.125 in radii. The 0.250 inch radius was broken into production during Block 40/42.

In 1993, aircraft on which ECP 1992 had been accomplished were beginning to re-crack. It was first believed that the crack tips were not being removed by the undercut, but was later shown not to be the case. One year later after several analyses, an aft fuselage strain survey and several more component tests, ECP 1992 was proven to be insufficient for pre-Block 40 aircraft. Two factors contributed: The original internal load distribution under predicted the bending moment at FS 479 by 12 percent and the vertical tail load spectrum had been under predicted. A fleet wide inspection was conducted for all F-16 pre-Block 40 aircraft. Inspection results revealed fleet wide cracking to an extent not anticipated. An improved bulkhead design along with a material change from 2124-T851 to 2097 aluminum lithium has proven to be sufficient to meet the service life requirements for today's usage environment.

Every USAF F-16 C/D aircraft is monitored through the Individual Aircraft Tracking (IAT) Program. This program enables the USAF to monitor the usage environment on each aircraft and project inspection intervals and service life. Based on today's usage environment, the Falcon Up Program should allow the F-16 to achieve its 8,000 hour service life. Should the usage environment change as before, additional structural modifications may be necessary to reach the de-desired service life.

8.0 ACKNOWLEDGMENTS

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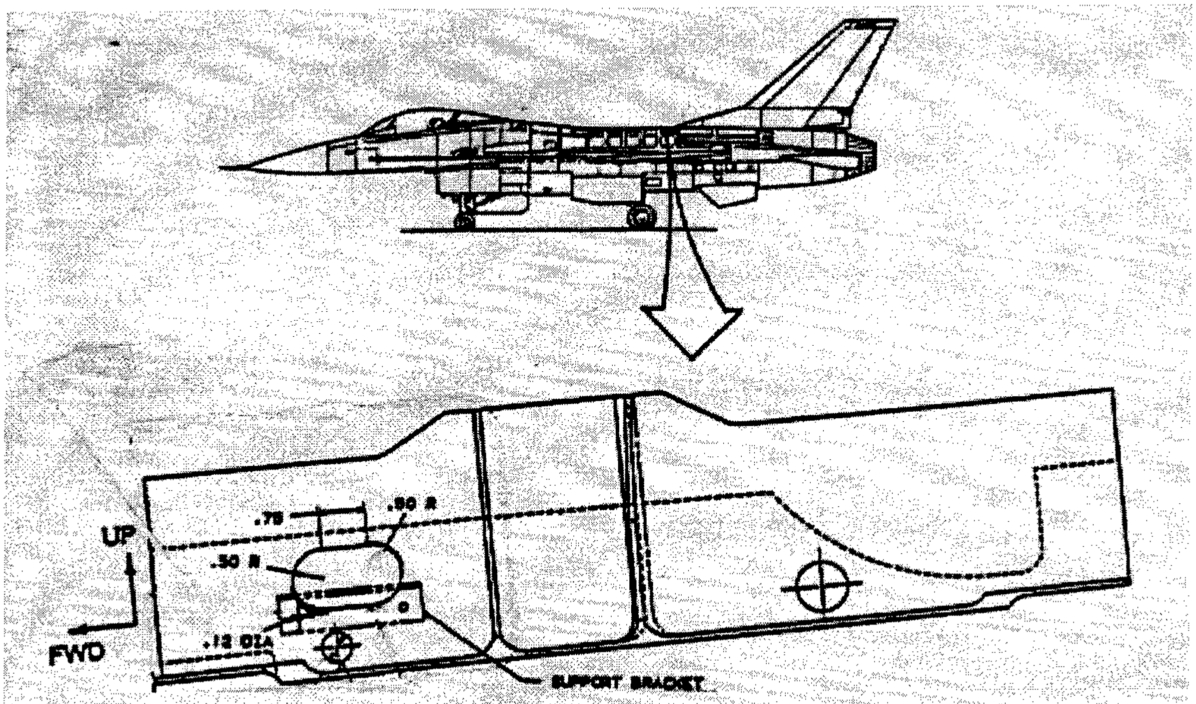
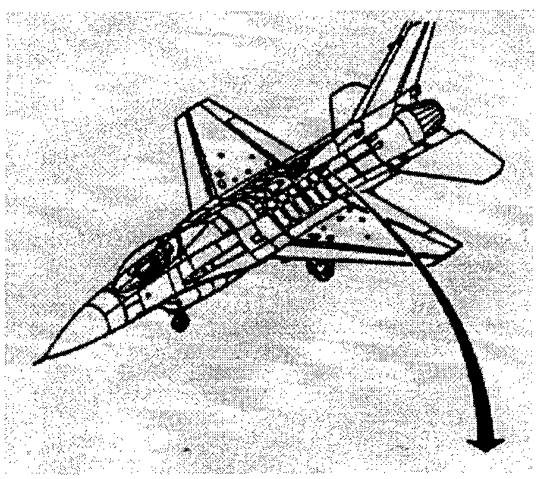


Fig 13



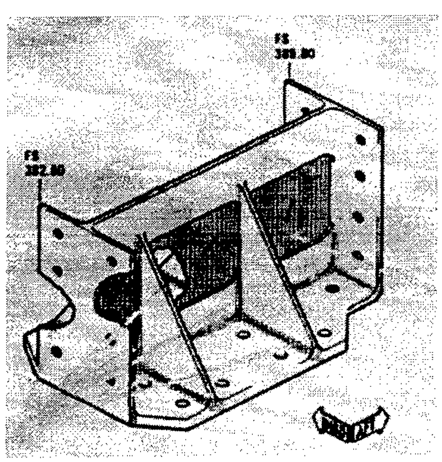


Figure 14

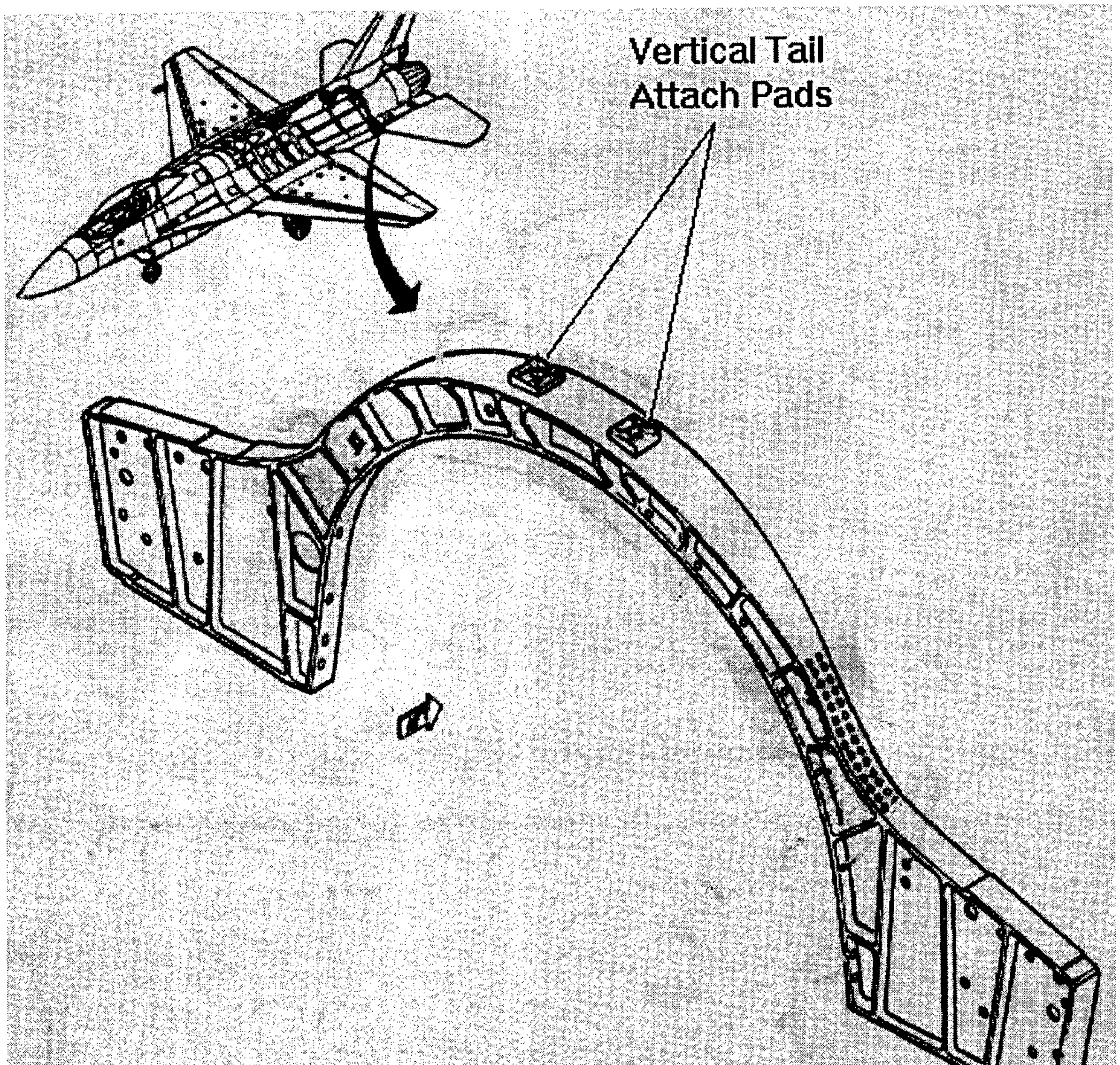


Fig 15

Jim Mills, Flight Controls Engr, F-16 SPO
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Joe Morrow, Fatigue & Fracture Engr, LMTAS
Aubrey Stratton, Structures Engr, LMTAS

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Coral Reach
USAF KC-135 Aging Aircraft Program

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1. Summary

1.1 Purpose
This paper is intended to serve as an aid to any aircraft manager beginning to develop or expand an aging aircraft program. It is based on the experiences of the USAF KC-135 aircraft program since the early 1990's as they have fought to gain recognition and support for the issues faced by that program.

1.2 Aging Aircraft Concern
CORAL REACH is a USAF program responsible for developing a GRAND STRATEGY for all age related issues on the C/KC-135 aircraft fleet. It is intended to ensure a logical, comprehensive and proactive program to sustain the aircraft until retirement. Today, the KC-135 aircraft have an average age of 38 years and many are expected to serve until the year 2040. Specific life extension efforts for the aircraft are dependent on the outcome of analyses and studies performed by the Integrated Product Team (IPT) incorporated under the CORAL REACH program. These analyses and studies focus on the technical, economic and safety aspect of the issues and become the basis for future actions to deal with the effects of an aging aircraft fleet.

1.3 Strategy
The CORAL REACH strategy is documented in an Aircraft Sustainment Master Plan (ASMP) which serves as an investment, repair and modification decision guide for the aircraft. With this plan, the USAF expects to maintain a safe aircraft yet keep the cost of ownership within reasonable bounds.

2. Background

2.1 Determination of Need
In the early 90's the USAF started serious planning for replacement of several older USAF weapon systems including the KC-135 tanker. It quickly became apparent that it would be impossible to financially afford to replace the large fleet of KC-135 tankers in the near term at the same time that other systems were being replaced. USAF planners needed a clear picture of how long the KC-135 tanker could be maintained as a viable part of the USAF fleet.

Some structural analysis had been performed in the 70's on the KC-135 that indicated a minimum safe flying hour life of 43,000 flying hours. However, the analysis did not define the maximum flying hour life nor degrade that projection to account for the effects of corrosion and wide spread fatigue damage on the structure. The need for this new consideration did not become apparent until the early 90's when structural tear down of retired aircraft highlighted the need for additional analysis. An additional analysis in 1995 showed that the KC-135 aircraft could theoretically fly to 70,000 hours. This would suggest a life beyond the year 2100 as shown in fig 2.1. However, no method was available to incorporate the effects of corrosion and wide spread fatigue damage. Technology was very weak and immature in these areas so the KC-135 program office established its own program to determine how these factors would degrade life projections for the KC-135 aircraft. Additionally, it was recognized that the systems on the aircraft would not last forever and their life span was unknown also. As a result, there was no readily available answer on true life expectancy for the aircraft.

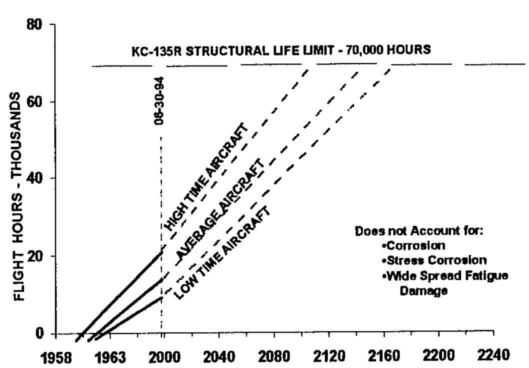

Figure 2.1

At the same time, corrosion effects had started showing up on major structural components during programmed Depot Maintenance (PDM). Few of the major structural items had been purchased in large quantities earlier in the life of the aircraft because there was no expectation that the aircraft would be flown for so many years. This led to delays in PDM schedules as well as unplanned expenses to procure expensive replacement parts.

Paper presented at the AGARD SMP Lecture Series on "Aging Combat Aircraft Fleets — Long Term Applications", held in Madrid, Spain from 7-8 October 1996; Pomezia, Italy from 10-11 October 1996; Atlanta, USA from 4-5 November 1996 and Brussels, Belgium from 22-24 January 1997, and published in LS-206.

Preliminary estimates indicated that the aircraft should be able to fly until at least the year 2040 if investments were made to restore the structure integrity where corrosion was an identified problem. In addition, selected systems like aircraft wiring were being replaced to restore system integrity in the aircraft electronics. This provided sufficient information to create a baseline for initial planning.

2.2 Initial Baseline

With this initial baseline, the USAF had a basis to start planning for a replacement tanker. However, the baseline did not provide sufficient details or accuracy to develop specific plans to actually achieve the goal of aircraft operation until the year 2040. It simply said that it was feasible if sufficient resources and system upgrades were invested in the KC-135. The next step was to gather data and develop a plan on how to maintain and upgrade the KC-135 such that a safe and affordable aircraft would serve the USAF until 2040.

Figure 2.2 provides a notional view of how a one for one aircraft replacement program might be accomplished at different spending rates. With a large fleet, it is obviously a very expensive program spread over a long period of time.

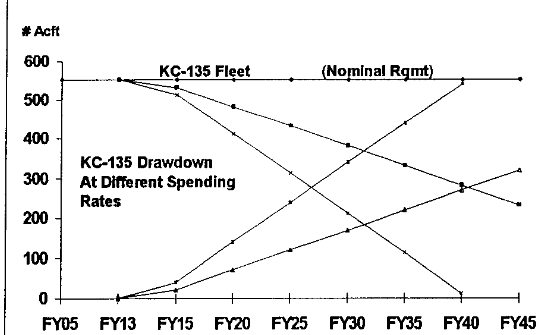

Figure 2.2

2.3 Program Initiation

The first step in this process was to make an assessment of the extent and effects of corrosion on the KC-135 aircraft. This stage of the process began in 1991 with a tear down and analysis of a retired EC-135 aircraft along with sections from other aircraft such as the B-52 and Boeing 707's. As the program developed, it became obvious that the effects of aging aircraft could not be restricted to the structural portion of the aircraft. The program was later expanded in August 1994 to add coverage for the systems on the aircraft. At that point, the program was given more formal recognition under the title CORAL REACH. It is the CORAL REACH program that is intended to provide the USAF with a GRAND STRATEGY on how to actually fly the aircraft until the year 2040. With those answers in hand, the USAF will also be able to make specific plans for procurement and phase in of a new tanker to replace the KC-135.

3. STRUCTURAL TEAR DOWN AND CORROSION ANALYSIS

3.1 Structural Tear Down Initiated

This phase of the effort occurred primarily in the 1991 to 1994 time frame. It centered on the tear down and preliminary analysis of an EC-135 aircraft which is a modified version of the basic KC-135 aircraft. The aircraft chosen had served in high corrosion areas of the world and was expected to contain examples of the types and locations of corrosion to be found on the KC-135 aircraft in general. Additional samples were taken from retired B-52 aircraft as well as retired Boeing 707 aircraft. Although the information to be gained would be valuable to other programs, the initial efforts were focused on specific issues related to the KC-135 rather than overall advancement of technology. Subsequently, the effort was joined by other governments agencies, contractors and academia seeking information in the same areas. However, the KC-135 program office and the USAF had questions to answer on the KC-135 and our efforts were focused on answers to those questions. The tear down and subsequent test and analysis was focused in the following areas.

3.2 Evaluation and identification of Non-Destructive Inspection (NDI) Equipment

No reliable method existed which would locate corrosion in hidden areas of the structure of an aircraft. If corrosion was suspected, invasive disassembly was necessary. This can lead to maintenance induced damage in areas that did not need disassembly. To address this problem, sections of the aircraft with known or suspected corrosion were cut out as specimens for later testing. Those specimens were taken to a lab where contractors were allowed to search for hidden corrosion with their NDI equipment. Many existing and new technologies were tested. After all testing was completed, the specimens were disassembled and the actual corrosion was mapped out and compared to the results from each contractor's testing. The results provided a basis for further testing using the best technologies.

Although the testing is still in progress, some technologies offer real promise. It is expected that prototype equipment will initially be developed for depot level use but expanding to field level use as the technologies mature. Eventually, equipment will be introduced into the maintenance environment as part of a routine inspection process.

3.3 Corrosion mapping

Although several areas of the KC-135 were known to be corrosion problems, no systematic mapping of the corrosion problem areas had been accomplished to date. It was critical that these areas be identified so that planning action could be established to inspect suspect areas and procure parts for repair. The tear down of the EC-135 aircraft offered an excellent opportunity to establish a baseline mapping of corrosion areas on the KC-135. Data collected from the actual tear down combined with data from the NDI testing on specimens taken from this aircraft provided the first real baseline of corrosion on the KC-135. This information was critical in the ability to establish inspection plans for the rest of the fleet as well as support decisions on parts to procure to support future maintenance actions.

3.4 Corrosion/Structural Integrity Testing and Analysis

The structural integrity for the KC-135 as well as all USAF aircraft has been based on the Aircraft Structural Integrity Program (ASIP) for many years. The ASIP is based on the Damage Tolerance Analysis (DTA) concepts. Although the ASIP and DTA concepts have an excellent foundation, they are based on pristine structure and do not include considerations for how corrosion may alter crack growth rates.

To address this problem, specimens from the tear down aircraft are undergoing a variety of testing to determine how corrosion affects structural fatigue life. It is desired to find out how crack growth is impacted as well as determine a quantitative way to measure corrosion such that it can be properly accounted for in analysis and maintenance actions.

3.5 Corrosion Growth Rate testing and analysis

The rate of corrosion growth in unique structures on the KC-135 is needed to establish maintenance plans and parts procurement. The tests are intended to identify the variables that influence growth rates such as type of construction, material combinations and environmental factors. To be useful, this information must be developed into a quantitative method that can be used to determine proper maintenance actions as well as planning for future maintenance and procurement of parts.

3.6 Repair History Data Base

One major shortfall in the ability to plan for the future was the lack of any integrated data files that contained data on the history for each aircraft. Aircraft selected for retirement were based primarily on knowledge that individuals had about selected tail numbers and general knowledge about the different series of aircraft. In addition, aircraft maintenance planned for accomplishment in PDM was based on a general knowledge of the aircraft rather than known problems for a specific aircraft. This led to surprises, unscheduled maintenance, and delays in production of the aircraft from PDM.

This concern is being addressed with the development of an integrated data base and data systems to capture historical data for each aircraft. The data base captures information such as replacement of critical structural components as well as other information that assists planning for future maintenance actions at depot level. This data will help avoid technical surprises and enable improved planning for future maintenance.

3.7 Corrosion Predictive Modeling

Prediction techniques for corrosion are weak at best. A reliable method is critical to successful execution of the USAF strategy to fly the KC-135 until 2040. Most structural components have procurement lead time of 18-24 months. Prior to contracting for the items, funding must be programmed years in advance to avoid a funding crisis. This means that a funding or maintenance crisis can only be avoided if the corrosion problems can be reliably predicted 5-6 years in advance. No such model exists today.

Several programs are underway in this area to establish models that can serve this need. However, there is no near term solution here. The results from the efforts described previously must reach a higher level of maturity before sufficient data will exist to support development of this model.

In addition to the focused efforts by the KC-135 program, there are many other efforts by other agencies that will be critical to overall success in advancing the technology of corrosion detection and prediction. As this information becomes available, it will be integrated into the planning action by the KC-135 program.

4. CORAL REACH

4.1 Need for a Formal Program

Although the initial KC-135 aging aircraft issues were focused on corrosion affects to the structure, it soon became evident that the aging aircraft issues extended to the entire aircraft including all the systems. The SPO had already initiated a rewire program that would replace old wiring on the aircraft in several phases. Also in place was a small program called CORAL UPGRADE. This program focused on the actions necessary to identify critical structural parts that needed to be stock listed and procured in sufficient quantities to support the PDM program. Although each of these programs were operated within the SPO, they were not integrated into a comprehensive program that addressed all the aging aircraft issues faced by the KC-135 program. To provide this integrated effort, the CORAL REACH program was established in August 1994.

4.4 Coral Reach Charter
The purpose of the CORAL REACH program was to integrate all the various aging aircraft issues and activities under one umbrella effort. To execute this program, an Integrated Product Team (IPT) was formed. The IPT had members of the SPO at the core but also had significant support and participation from numerous other agencies such as:

- Using Major Commands
- Other Dept of Defense Services
- Military and commercial laboratories
- Contractors
- Federal Aviation Administration (FAA)
- National Aeronautical and Space Administration (NASA)
- Academia

The IPT has a charter to

*"Develop, coordinate and execute an overarching **GRAND STRATEGY** for aging aircraft related issues to insure a logical, comprehensive and proactive aging aircraft program to sustain the KC-135"*

4.3 Focus
The initial focus of the IPT was to develop and document a plan to address the issues. The plan was initially published in June 1995 as the C/KC-135 Aircraft Sustainment Master Plan (ASMP). Periodic updates will allow for integration of new information to upgrade the data and planning factors contained in the plan. The ASMP is fully focused on issues associated with the KC-135 aircraft. Since it is not available for general distribution and would have little value outside the KC-135 program, a conceptual "walk through" of the ASMP is provided in the following paragraphs to assist other organizations in the establishment of their own master plan.

5. Aircraft Sustainment Master Plan (ASMP)

5.1 Introduction.
This segment provides the *purpose, background* and *assumptions* used by the CORAL REACH IPT. The *purpose* section provides the reader with a statement of the goals, objectives and scope of CORAL REACH along with how the plan is organized. The *background* section provides information on aircraft fleet operations and condition along with a description on the methodology used by the CORAL REACH IPT to conduct the analysis contained in the plan. The final section on *assumptions* provides an understanding on what type of assumptions were made by the CORAL REACH IPT in the course of preparing the ASMP.

5.2 PDM Improvement, Aircraft Availability and Cost per Flying Hour
This segment provides a background of selected metrics critical to the KC-135 aircraft along with short and long term actions planned to address those metrics. The maintenance concept is discussed with specific attention to how it impacts the availability of aircraft to the user. It also discusses the cost per flying hour to maintain the desired levels of aircraft availability. To meet these needs, various action plans were established. The plans cover issues such as parts usage and forecasting, safety margins, cost analysis, depot and field tasks, inspections and computer systems needed to manage the information associated with these issues.

5.3 Structural Integrity
Since the bulk of work performed to date has focused on the structural aspects of aging aircraft, this segment contains the most in-depth coverage of issues related to the KC-135. It provides an explanation of current structural integrity concepts applied to the KC-135 under the USAF Aircraft Structural Integrity Program (ASIP). The limitations of the current ASIP to cover corrosion are also discussed and how those limitations affect the KC-135 aircraft.

Each area of the KC-135 structure is evaluated with both short and long term actions identified to address shortfalls. The short term actions concentrate on items that need immediate attention to address known deficiencies impacting current maintenance at depot and field level. The long term actions focus on additional studies and analysis required to develop proactive plans to avoid "surprise" maintenance issues in the future. These include actions such as:

- Fatigue studies
- Corrosion growth rates
- Corrosion forecasting models
- Corrosion mapping techniques
- Data system requirements
- NDI technology
- Failure Modes and Effects Analysis

5.4 System Integrity
As aging aircraft issues were evaluated, it quickly became apparent that the "systems" on the aircraft also suffer from the effects of aging as much as the structure. Problems ranged from diminishing sources for parts and repair to degraded performance from aged components. This effect had been recognized earlier with the wiring system on the aircraft. In response, a rewire program is in progress that will eventually replace all the wiring in several phases. The initial phase to replace all the wiring for "flight critical" systems is essentially complete. However, this program only touches one of many systems that must be replaced or upgraded to achieve the goals of CORAL REACH. Current action is focused on additional systems such as circuit breakers and the fuel systems and components.

This segment of the ASMP addresses this issue with a background of the systems on the aircraft and provides short and long term action plans to resolve concerns.

The background section describes the historical approach to the issue along with an explanation of the limitations and deficiencies of the approach. It also describes the revised approach using a Functional Systems Integrity Program (FSIP) approach. This approach is much more structured with an analysis of each system to look for weaknesses related to aging effects. The initial results of the analysis led to a group of short term actions to address immediate concerns. The long term issues are also covered with a discussion of how to fully implement the FSIP philosophy to allow for proper planning for the future.

5.5 Data Collection
On major shortfall discovered when establishing the CORAL REACH program was the lack of data and data systems that would assist in decision making. Although some bits and pieces of information could be found in various sources, much of it was in manual records and no integration of the data had ever occurred. This situation was another fact of aging aircraft. Many newer aircraft system were born in an era where data systems were part of everyday life thus plans had included data automation and data collection from early in the life of the aircraft. This is not true for most of our aging aircraft of today.

The ASMP addresses this shortfall with a description of the existing data systems along with their limitations and deficiencies concerning aging aircraft. Short term action plans focused on capturing existing manual records prior to their disappearance. Additionally, the capture of critical maintenance information from current activities was initiated. For long term actions, various data system are being designed and developed to serve as a host for this information along with methods to retrieve the information suitable for decision making associated with aging aircraft issues.

5.6 Recommendation Task Summary Matrix
This segment captures all the actions identified elsewhere in the ASMP and summarizes them into several categories. It is designed as an aid to management to ensure that tasks are readily identifiable in a single location plus allow for segregation according to funding situations. Categories chosen are:

- Category 1 - Current activities
- Category 2 - Action previously funded or initiated through separate action
- Category 3 - Actions which require action or funding through the year 2001
- Category 4 - Actions required beyond the year 2001

5.7 Budget Projections
As with any program initiated inside the lead time for funding through the normal budget process, the CORAL REACH program has suffered from funding problems. This segment summarizes all the funding required to execute the plans contained in the ASMP. Unfortunately, the funding required is very significant and difficult to obtain in time to execute the CORAL REACH program as desired. However, the information is documented here to support decision making.

5.8 Conclusions/Cost of Ownership
This segment provides some data and conclusions concerning the cost to maintain the KC-135 aircraft into the 2040 time period. Although no solid data exists to make accurate projections, a look at historical cost combined with estimates about the future does provide some insight on the most probable future cost of ownership. One example of ownership cost is a cost per flying hour projection as shown in figure 5.1. The SPO is using a "reasonable bounds" approach to provide USAF leaders with an estimate of future cost. This provides an upper and lower bounds along with a most probable cost.

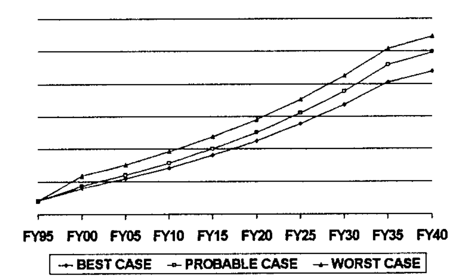

Figure 5.1

5.9 Appendices
This last segment is used to contain various reference materials including descriptions of analysis techniques, references, acronym list and primary points of contact. It can include any sort of data or information needed to support other segments of the ASMP.

6. Lessons Learned and Recommendations
Establishment of the CORAL REACH program has been a significant learning experience for the KC-135 SPO. Although many of the issues have factors unique to the operations within the USAF, most have a generic issue associated with them that could be faced by any organization attempting to establish a similar program. To assist other organizations in their planning, the following is offered as a set of lessons learned with some recommendations on how to avoid some of the problems.

6.1 Too Much Focus on Technical Issues
The roots of CORAL REACH go back to several engineering efforts focused on corrosion affects on

aircraft structures. Although these programs were vital to determination and resolution of technical issues, they had a tendency to create the illusion that those efforts encompassed all aspects of the problem and solutions. They were definitely a critical part of the future plans to be created under CORAL REACH but they did not capture all aspects of the problem such as the aging systems on the aircraft. To do that required a program manager with an Integrated Product Team (IPT) who could integrate the technical information into a STRATEGIC PLAN that would procure parts and equipment plus develop budgets, maintenance concepts and plans for the future.

6.2 No Clear Statement on Life Expectations
Since the KC-135 aircraft had served faithfully for over 30 years and no obvious limitations existed, the USAF had no clear stated objective on what life was expected from the KC-135 aircraft. When aging aircraft issues started becoming apparent, the questions began to arise on how long can the aircraft be flown and what would be the cost. Additionally, it quickly became obvious that other USAF aircraft were experiencing aging aircraft problems also. This created a dilemma on which aircraft should be replaced first within a very constrained budget.

Since no life target had been set previously, there was no game plan established to reach the target. This created a planning void that would later be filled by CORAL REACH. If a target had been defined earlier, it would have created a baseline of information to work with thus reducing the CORAL REACH start up problems.

6.3 Technical Surprises
There was little advance warning concerning the magnitude of problems to be addressed by an aging aircraft. Maintenance concepts had basically focused on repairing what was broken with historical data as a primary indicator of things to come. Many items on the aircraft (like primary structural components) had never been purchased in the past 30 years. Where some had previously been purchased, it was typically in small quantities (2 -3 each) as insurance only. It was never envisioned that the aircraft would stay in service long enough to consider replacement of these major structural components in significant quantities.

An additional complication that had not previously been considered was the fact that much of the aircraft was made from alloys popular in the 50's. These alloys have proven to be very vulnerable to stress corrosion effects and that problem is now showing up after 30+ years in large numbers for major structural components.

These factors started coming together in the early 90's creating significant technical surprises for the KC-135 SPO. As a result, inspections were expanded. This further compounded the technical surprise situation because the increased inspections served to uncover additional corrosion problems. Since none of this had been anticipated, parts to support maintenance and funding to buy the parts was not readily available.

This served to point out that it is never to early to start an aging aircraft program. Even aircraft programs just being started should plan to capture information that will become critical in later years as the aircraft ages.

6.4 No Infrastructure Existed to Support Planning
A major stumbling block was the absence of any USAF infrastructure to address aging aircraft issues. The entire management and funding structure for aircraft management is based on either purchasing a new system or sustainment of an existing system based on historical data. There were no funding concepts, policy guidance, senior level awareness of the issues or support structure for development of an aging aircraft program. As a result, the entire CORAL REACH effort became an educational process for the USAF as well as academia, contractors and other participating governmental agencies.

Funding was a significant problem in this environment. First we had to convince the budgeting process that the aging aircraft issues were real and that the budget data was reasonable. Additionally, there were no established budget categories that allowed the program cost to be consolidated into a few line items. Each effort often competed for funding as a stand alone effort leading to program disconnects when different portions of a program received incomplete funding. This was further complicated by the fact that we were inside budget lead times. Any funding we received was at the expense of another program somewhere in the USAF.

The key here is to establish "program" funding where possible. This will avoid the chase for pieces of money leaving you with an incomplete program.

6.5 Missing Data and Data Systems
The KC-135 is over 30 years old and much of the information needed to support aging aircraft issues was collected manually. During that period much has also been disposed or lost.

An additional issue is due to the lack of any established data system with a focus on aging aircraft issues. This prevented that capture of additional information that would be useful to decision making for an aging aircraft.

To address these deficiencies, the KC-135 SPO initiated several efforts to capture all the manual data to be found that would assist aging aircraft analysis or decision making. Additionally, data systems are planned to host the data and provide means to analyze the data for decision making.

6.6 Diminishing Margins of Safety
Although the KC-135 aircraft is a very robust aircraft, there were initial concerns about safety that could not be resolve in a timely manner. As a result, our initial analyses were focused on potential safety areas. Fortunately, the analysis indicated that there were no near term safety issues. The issues were mainly economic, maintainability and fleet availability concerns. However, failure to act now would eventually lead to safety concerns as corrosion and other aging aircraft issues would cause diminishing margins of safety over time. Each of these concern areas now has specific actions underway to repair, replace of modify the area to eliminate any near term concerns. It should be noted that the areas include functional systems on the aircraft in addition to the structural concerns most managers are expecting.

6.7 Functional System Integrity Program - A Missing Link
Traditionally, an aircraft life is measured in structural terms. As a result, technology has focused in this area for many years. However, the reality of aging aircraft has forced a new player to the surface where systems are becoming a life limiting factor as we fly aircraft far beyond initial expectations.

In the USAF, system integrity has traditionally relied on reliability and maintainability (R&M) programs that have not considered the true effects of aging systems. When R&M fell to unacceptable levels, upgrades or replacements were incorporated on a selected basis. Major upgrades were seldom made on "old" systems because pay back was difficult to achieve in the remaining life of the aircraft. However, the remaining life of our older systems are now often much longer than previous experience. This is forcing us to rethink our philosophy for system improvements.

In the USAF, the new philosophy is developing around a concept called the Functional System Integrity Program. This program supplements the existing R&M concepts with analyses that focuses on aging aspects of functional systems on the aircraft. Although this concept is gaining recognition, it has not yet been institutionalized in the USAF. This void makes it difficult to budget for funding to achieve the desired benefits of the program.

Despite this shortfall, the KC-135 program has pressed forward with initiatives for the KC-135 that incorporate the tenants of the FSIP. The FSIP is intended to be a proactive program to ensure safety and durability for the aircraft systems over the remaining life of the aircraft. This requires a systematic analysis of each system to identify single or multiple failures that could jeopardize the safety of the flight crews and/or the aircraft. This type of analysis is normally accomplished in the beginning for any new aircraft system but is seldom re-accomplished in a systematic way over the remaining life of aircraft. This leaves opportunity for usage or other unknown changes to occur which effectively invalidates the original analysis.

This is further compounded with the effects of an aging aircraft when systems are expected to serve far beyond original expectations. However, with a systematic process such as a FSIP, all these effects can be analyzed and solutions can be planned which provide for safe, effective and maintainable systems for the full life span of any aircraft system.

7. Conclusion
Every aircraft manufacturer and manager needs to consider the effects of aging aircraft from the beginning. If that was not done initially, start immediately. This need is driven by the reality that the expense of aircraft replacement drives the users to retain existing aircraft far beyond the life originally expected or desired. This is becoming painfully evident to most aircraft fleet managers every day.

Even though the KC-135 SPO has established a firm baseline for future action, financial challenges abound. Every aircraft manager should expect the same as limited finances must be balanced between maintaining the existing fleets while attempting to procure replacement fleets.

Time is a formidable enemy. There will be considerable pressure to defer action for many reasons. However, this will only compound the problems of the aging fleets. **<u>Now is the time for action!</u>**

8. Abbreviations and Definitions

ASIP - Aircraft Structural Integrity Program
ASMP - Aircraft Sustainment Master Plan
CORAL REACH - Name assigned to the USAF program for action on aging aircraft issues on the KC-135 aircraft
CORAL UPGRADE - Name assigned to the USAF program focused on stock listing and procurement of structural components for the KC-135 aircraft. It is a sub element of the CORAL REACH program.
DTA - Damage Tolerance Analysis
FSIP - Functional System Integrity Program
IPT - Integrated Product Team
KC-135 - USAF designation for the largest tanker fleet of aircraft in the world
NDI - Nondestructive Inspection Program
PDM - Programmed Depot Maintenance
R&M - Reliability and Maintainability
SPO - System Program Office.
USAF - United States Air Force

REPAIR/REFURBISHMENT OF MILITARY AIRCRAFT

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1. INTRODUCTION

The military and commercial aircraft fleet in the United States and throughout out the world contains a high proportion of aging aircraft. This percentage appears to be growing due to the reduction in the defense budget and the political environment throughout the world. There is tendency to use these aircraft beyond their original design life, and, in addition, the demand on the performance of these aircraft has been increasing due to higher pay-load, severe spectrum and extended service life requirements. Maintaining the airworthiness of these aircraft is of great concern to the regulatory authorities. The reduction in budget requires that the maintenance cost of these aircraft be kept to a minimum and at the same time assure the safety of flight. The advances in the structures, materials and processes, manufacturing and repair technologies provide opportunities to achieve the performance goals of these aircraft and at the same time reduce the support requirements and thereby reduce the life cycle costs.

This paper discusses structural life enhancement through prestressing techniques such as cold working, shot peening, laser shock processing, etc. The state-of-practice methods of repairing metallic and composite structures are outlined. Advanced repair methods such as composite patch repair of cracked metallic structures are shown. Finally, improved properties of advanced metallic materials are shown and their in-service applications to spare parts is discussed.

2. STRUCTURAL LIFE ENHANCEMENT

The life of an aircraft structure depends on the applied stresses, environments, structural details and the material of the structural component. Under certain loading and environmental conditions a crack may initiate and propagate in a metallic structural component. Depending on the structural details, the crack may result in a catastrophic failure or costly repairs. A logical preventive method is to retard the initiation and growth of the cracks by pre-stressing so that the cracks do not result in catastrophic failure before the useful life of the structure. In certain cases this may not be feasible and a structure may have to be repaired to meet the useful life requirements. In addition, the in-service damage in both metallic and composite structures frequently requires repairs so that the structure is able to carry the required load. The structural life enhancement techniques by prestressing and repairs are summarized in Figure 1.

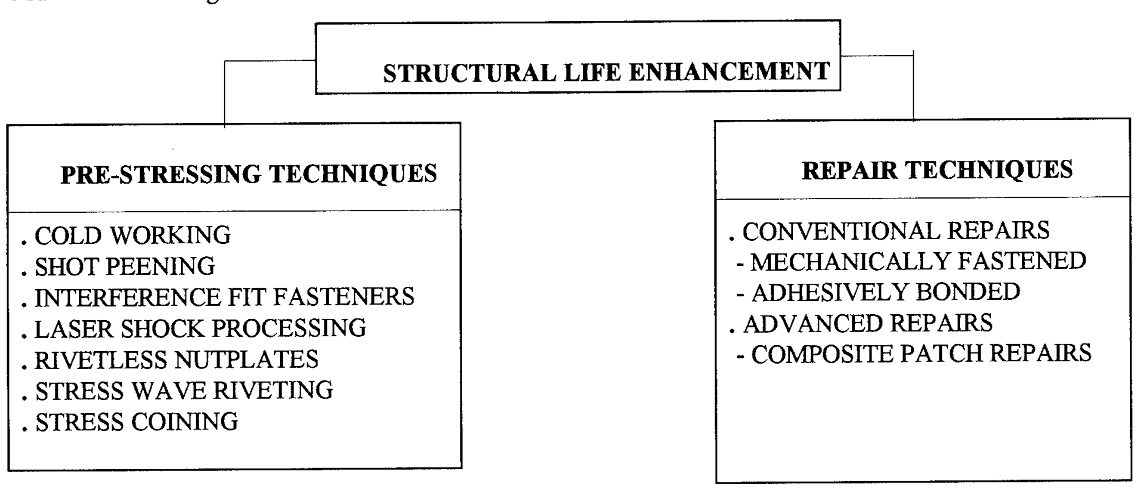

Figure 1. Life Enhancement Techniques

Paper presented at the AGARD SMP Lecture Series on "Aging Combat Aircraft Fleets — Long Term Applications", held in Madrid, Spain from 7-8 October 1996; Pomezia, Italy from 10-11 October 1996; Atlanta, USA from 4-5 November 1996 and Brussels, Belgium from 22-24 January 1997, and published in LS-206.

2.1 Life Enhancement Through Pre-stressing Techniques

In this technique a residual compressive stress field is created at the locations such as holes where cracks are likely to initiate. Subsequent inflight loads have to overcome the compressive stresses in order for the crack to initiate and propagate. Some prestressing techniques have been fully developed while others are still in the development stage and have shown good promise to enhance structural life. The application of these techniques to in-service aircraft are shown in Figure 2. The figure also shows the locations where these techniques are applied (e.g. whether the technique can be used at the manufacturing line, depot or field). The analysis methodology that can be used for life predictions is also shown in the figure. The level of verification testing required for successfully implementing the technique is also given in the table.

PRE-STRESSING TECHNIQUE	IN-SERVICE APPLICATIONS	LOCATION WHERE PERFORMED	ANALYSES METHODS	REQUIRED TESTING
COLD WORKING	T-38, F-5, F-16, JSTARS F-18, F-111, C-141, 747	MANUFACTURING LINE, DEPOT AND FIELD	EQUIVALENT INITIAL FLAW(EIF), FATIGUE LIFE FACTOR(FLF)	MINIMUM
SHOT PEENING	T-38, F-5, F-18, F-14, 737,747,C-130,B-1	MANUFACTURING LINE, DEPOT AND FIELD	EIF, FLF	MINIMUM
INTERFERENCE FIT FASTENERS	T-38, F-5, F-18, 747	MANUFACTURING LINE, DEPOT AND FIELD	EIF, FLF	MEDIUM
LASER SHOCK PROCESSING	NONE KNOWN	MANUFACTURING LINE	DEVELOPMENT REQUIRED	SUBSTANT-IAL
RIVETLESS NUTPLATES	F-22	MANUFACTURING LINE, DEPOT AND FIELD	EIF, FLF	MEDIUM
STRESS WAVE RIVETING	F-14, A6E	MANUFACTURING LINE AND DEPOT	EMPIRICAL	MEDIUM
STRESS COINING	F-18, DC-8, DC-9, DC-10	MANUFACTURING LINE AND DEPOT	EMPIRICAL	MEDIUM

Figure 2. Life Enhancement Techniques Applications

2.1.1 COLD WORKING

Cold working is a process in which a mandrel is pulled through a hole so as to produce radial expansion of the hole and create residual compressive stresses around the hole. The amount of compressive stresses produced around the hole depends on the extent of radial deformation at the hole. Typical compressive stresses produced by different percentages of the applied expansion are shown in Figure 3. The figure shows the variation of residual hoop stress normalized with the yield stress of the material as a function of the distance from the edge of the hole for three different applied expansions of 3.1, 4.9 and 6.3 percent.

A significant amount of experimental and analytical work has been done for the cold working method of life extension technique (References 1-11). Two common cold working methods used are-1) Split sleeve which uses a solid mandrel with a sleeve split longitudinally to facilitate placement over the mandrel, and 2) Split mandrel without a sleeve. The cold working method using a split sleeve, used by Fatigue Technology Inc. (Ref. 1), is shown in Figure 4. The split sleeve method has been used in a number of in-service and new production aircraft for enhancing the life. The effect of cold working on fatigue lives of 7075-T-6 and 2024-T3 aluminum lugs (Ref. 3), using the split sleeve method, is shown in Figure 5 for 3 percent hole expansion, indicating significant increase in fatigue lives.

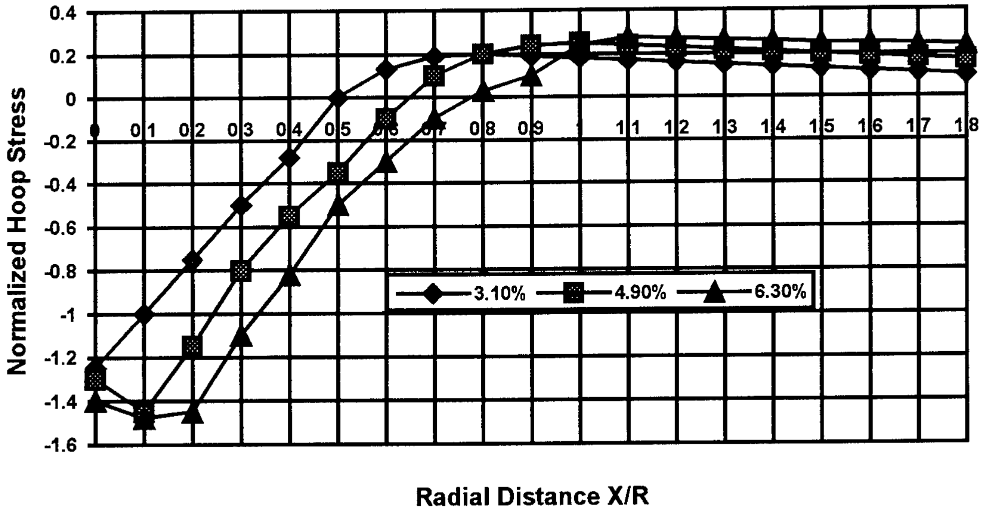

Figure 3. Residual Hoop Stress at the Hole Due to Cold Working With Different Percentages of Applied Expansion

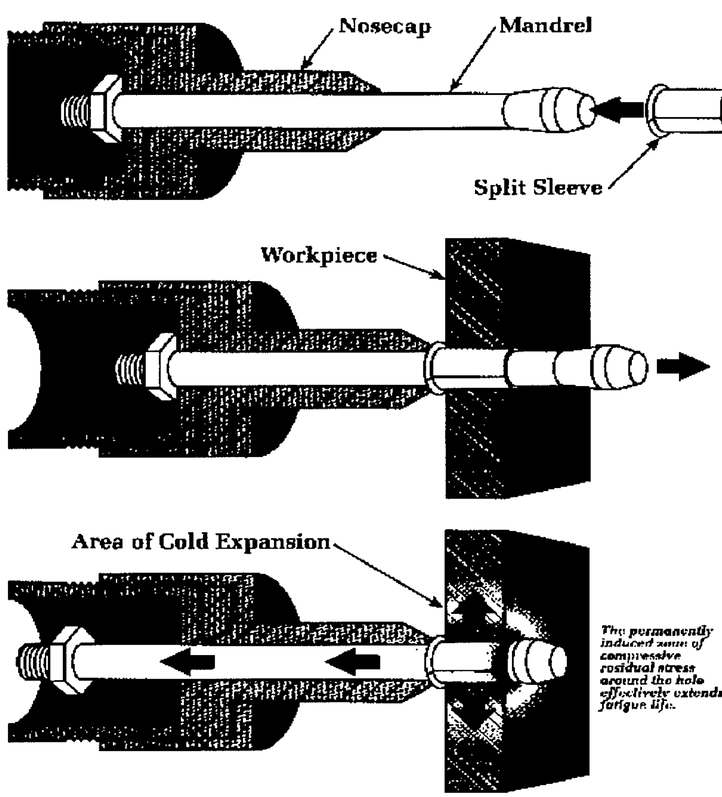

FIGURE 4. Cold Working Process Using Split Sleeve

The fatigue life improvements shown in Figure 5 are for constant amplitude loading. The life improvements under spectrum loading are generally smaller compared to constant amplitude loading due to possible relaxation of residual stress 'under service loads. Under spectrum loading, fatigue life improvement factors of 3 to 5 were found for 2024-T3 and 15 for 7075-T6 aluminum lugs in Ref. 3. Improvements of a similar nature for flight simulated loading have been reported for mechanically fastened joints in Ref. 9 and 2024-T3 open holes in Ref. 10. These test results indicate that relevant testing under spectrum loading must be done for any new life improvement technique and that constant amplitude fatigue testing is not inadequate.

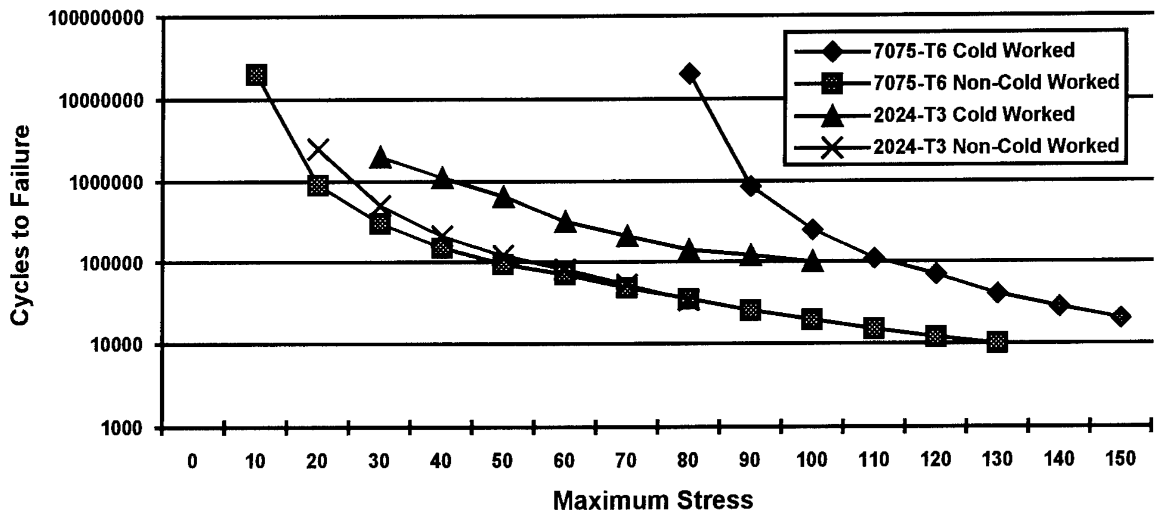

Figure 5. Effect of 3% Hole Expansion on S-N Curves of 7075-T6 and 2024-T3 Aluminum for 3 % Hole Expansion

The effect of pre-crack on the crack growth life of 7050-T7 aluminum has been investigated in Ref. 11 for constant amplitude loading. The specimens that contained pre-cracks of lengths of 0.02, 0.05, 0.08 and 0.12 (0.5, 1.25, 2.0 and 3.0 mm) were tested under constant amplitude loading with R=0.1 and a maximum stress of 30 ksi (207 MPa). The cracks in cold worked specimen were introduced prior to cold working. The retained expansion at the hole was about 2.2 percent. The upper and lower bounds of the fatigue life data for cold worked and non-cold worked holes are shown in Figure 6. The figure shows that the improvement in fatigue life depends on the initial crack length. Shorter crack lengths providing a larger increase in life.

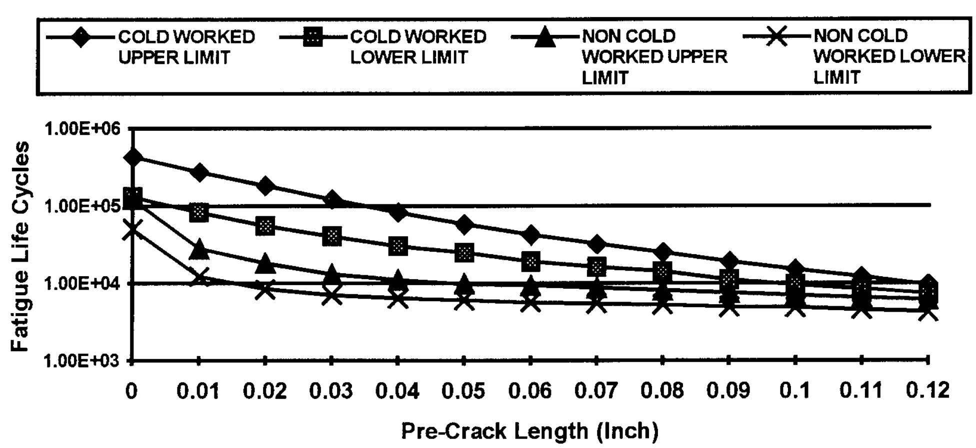

Figure 6. Effect of Pre-Crack Length on Fatigue Life of 7075-T7 Aluminum

2.1.2 SHOT PEENING

In this life enhancement technique a compressive layer is produced on the surface of a finished part by impacting small metallic, glass or ceramic beads (References 12-15). The intensity with which these beads are impacted depends on the material being peened and the amount of compressive layer desired. A structural component can be shot peened by passing the component through a machine or by shot peening with a hand held unit. A typical residual compressive stress field produced by two different levels of shot peening is shown in Figure 7.

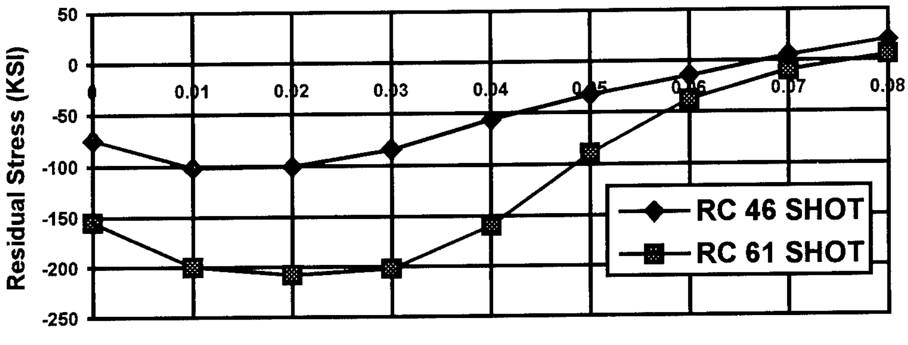

Figure 7. Residual Stresses at Various Depths of a Specimen Due to Shot Peening

Reference 12 studies have shown significant increase in the fatigue life of 7075-T6 aluminum subjected to reverse bending fatigue. Studies reported in Ref. 16 have shown increase in the stress corrosion cracking resistance of 7075-T6 aluminum after shot peening. Reference 14 studies have shown an increase in the fatigue life of 4340 steel after multiple shot peening cycles. Studies conducted in Ref. 15 have shown significant increases in the fatigue lives of several steel and aluminum alloys.

2.1.3 INTERFERENCE FIT FASTENERS

In this method an increase in life is achieved by creating compressive stresses at the hole through the radial expansion. The radial expansion is produced by using a fastener whose diameter is greater than that of the hole. The extent of life extension depends on the amount of interference. Greater benefit in life improvement is observed with increased interference (Ref. 17-18). Test results from Ref. 18 for three different levels of interference fit, obtained on specimen subjected to fighter spectrum loading, are shown in Figure 8 along with the test data for open hole. The figure shows plots of crack length as a function of flights. The figure shows that the fatigue crack growth life is highly dependent on the level of interference fit and it is possible to get large increases in fatigue lives.

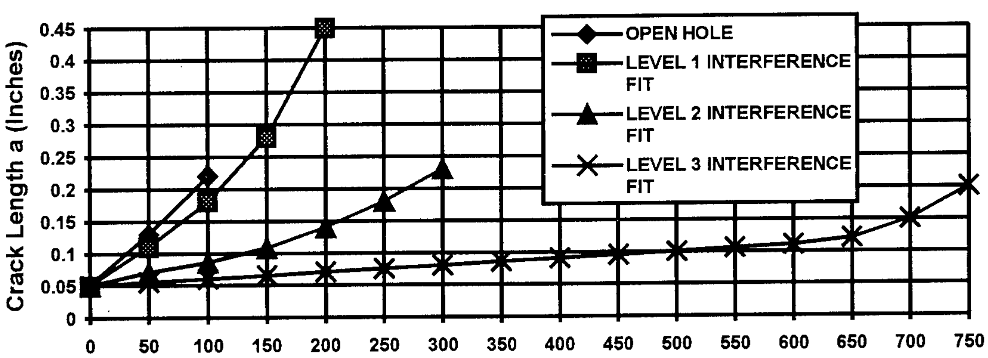

Figure 8. Crack Growth Behavior of Through Cracks from Interference-Fit Fastener Holes for Various Levels of Interference in 2219-T851 Aluminum Under Fighter Spectrum Loading

2.1.4 Laser Shock Processing

This process involves shooting a laser pulse onto the surface of a part that is covered with black paint and a transparent overlay. The vaporization of the paint sends a shock through the material causing a change in the microstructure of the material with an increase in the hardness and a residual compressive layer as shown in Figure 9. Most of the work in this area has been done by Battelle and the application has been limited to testing only. The testing has shown an early initiation and reconfiguration of cracks. The net result was a substantial reduction in crack growth rate for aluminum alloys. The process was found to be more effective in thin sections (0.125 inch) as compared to thick sections (Ref. 19-20). Extensive testing is required to qualify the process.

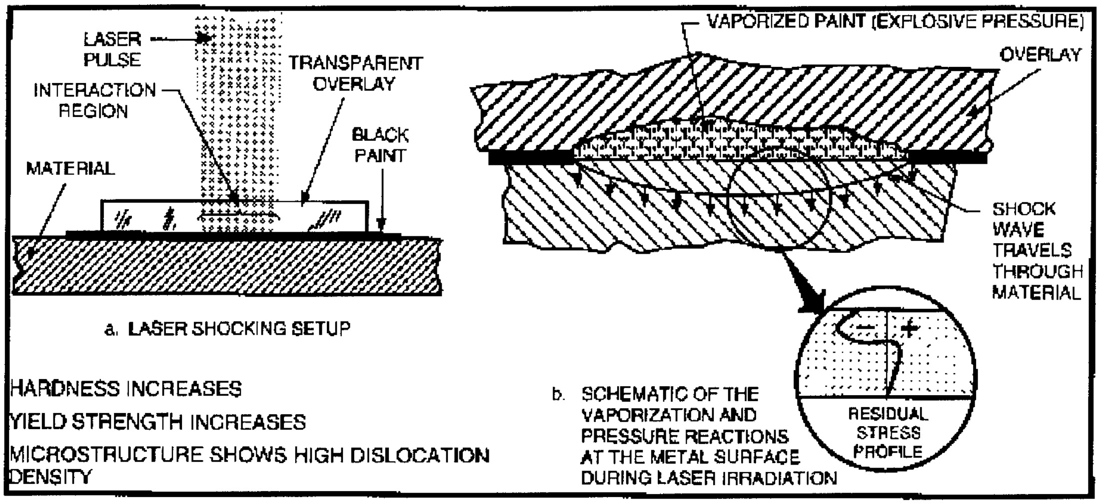

Figure 9. Laser Shock Process

2.1.5 Rivetless Nutplates

In this process, a nutplate integral with a bushing is cold worked into a fastener hole during the installation of a rivetless nutplate. The nut is caged on to the installation hardware and thus there is no need to rivet on the nutplate. The absence of the rivet hole reduces the stress concentration at the nutplate hole and also provides the benefit of cold working. This process has been developed by Fatigue Technology Inc. (Ref. 21). Figure 10 shows fatigue behavior of conventional nutplate and ForceTec nutplate, indicating significant improvement in fatigue life with ForceTec nutplate.

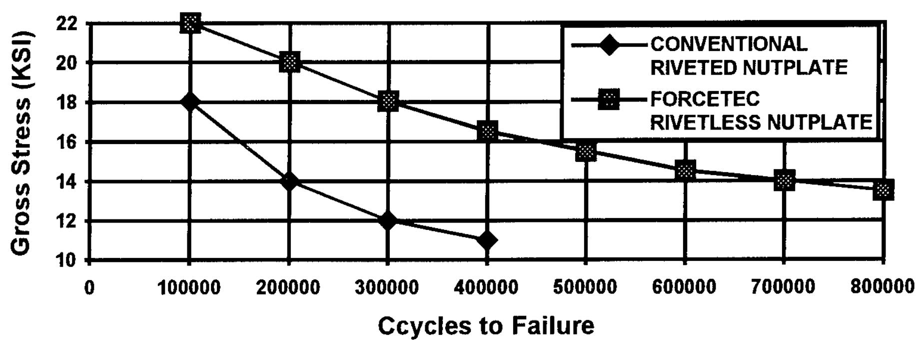

Figure 10. Comparison of Fatigue Lives of 1/4 Inch ForceTec Rivetless Nutplate and Conventional Riveted Nutplate in Low Load Transfer Conditions

2.1.6 Stress Wave Riveting

The Stress Wave Riveting (SWR) method was developed by the Grumman Corporation and uses a rivet that is impacted with a high velocity stress wave while being installed. The energy in the process causes the rivet to expand the hole and there by creating a residual compressive zone similar to that created by cold working. This method has a similar benefit in life extension as an interference fit fastener (Ref. 22). This technique has been used on F-14 bulkheads and A6-E wing covers. The technique (Ref. 22) is defined either as low (0.004 inch (0.1 mm)) or high (0.008 inch (0.2 mm)) interference. For open holes, the test results of Ref. 22 have shown the life extension by high interference same as for cold worked holes. Significant amount of testing and tooling development is required to make this technique as an industry acceptable practice.

2.1.7 Stress Coining

This life enhancement technique was developed by the McDonnel Douglas Aircraft Company (MDAC) and has been primarily used on their aircraft. Four different methods of stress coining are used. In radius stress coining, a highly polished 0.03 inch (0.8 mm) radius is cold worked around the edges of the holes and slots in material with thickness up to 0.188 inch (4.8 mm). In pad stress coining, a recess of approximately 0.004 inch (0.1 mm) is impressed in the surface material surrounding a hole or a slot in thickness of 0.188 inch (4.8 mm) or greater. Expansion pin stress coining method is similar to sleeveless cold working in which a lubricated pin is pushed through an undersized hole and expands the hole to its final diameter. In the ring pad stress coining method, two dies, having diameter greater than the diameter of the hole to be cold worked, are simultaneously compressed around both sides of the hole. This process causes a recess beneath the dies and leaves a ring pad adjacent to the hole. The ring pad coin method has shown a fatigue life improvement of 2.0 over a non-cold worked hole. The analysis of residual stresses produced by this technique is reported in Ref. 23. The application of this technique to other aircraft will require further testing and development of the process.

2.2 Life Enhancement Through Repairs

Structural life enhancement through repairs for in-service fatigue, corrosion and foreign object damage (FOD) has been well established for metallic aircraft. With the increasing use of composites for improved structural efficiency, these methods have been developed for composite materials. However, there are basic differences between the damage types and their behavior in composite and metallic materials (Ref. 24-26). Figure 11 shows a comparison of typical metal and composite fatigue behavior under fighter aircraft wing spectrum loading. The data are plotted for each material's most sensitive fatigue loading mode, which is tension-dominated (lower wing skin) for metals and compression-dominated (upper wing skin) for composites. The figure shows that composite fatigue properties are far superior to metal fatigue properties.

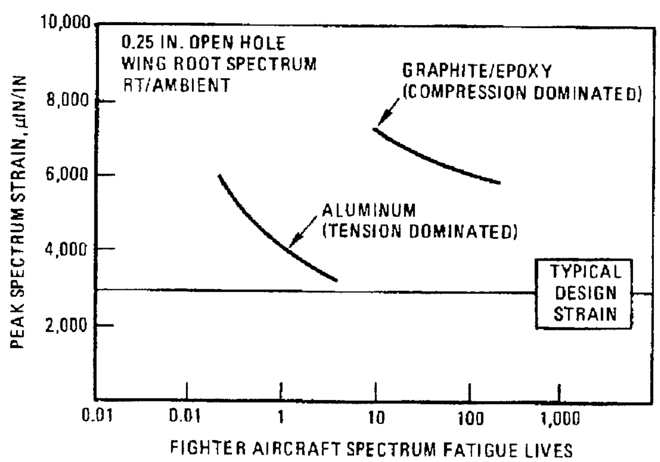

Figure 11. Comparison of Fatigue Behavior of Metallic and Composite Materials

One of the most important design consideration in the design of composite structures is the in-service impact damage. Impact damage can occur during ground handling and also during landing, take-off, and in-flight due to foreign object damage. Impact damage may be caused by hard objects (e.g. tool drops and runway debris) and soft objects (e.g. bird impacts that occur at low altitude during take-off and landing). The impact damage caused by tool drops, etc. is termed as low velocity damage. Studies have shown that considerable reduction in compression strength may occur due to low velocity damage that is not visually detectable on the impacted or other external surfaces. The non-visual damage may cause internal damage in the form of delaminations between plies, matrix cracking, and fiber breakage. The longitudinal cross-section of an impact damaged panel is shown in Figure 12. The damage due to impact is influenced by the factors such as laminate material properties, size of the laminate, support conditions, substructure, impactor size and shape, impactor velocity, impactor mass, impact location, and environment (Ref. 27).

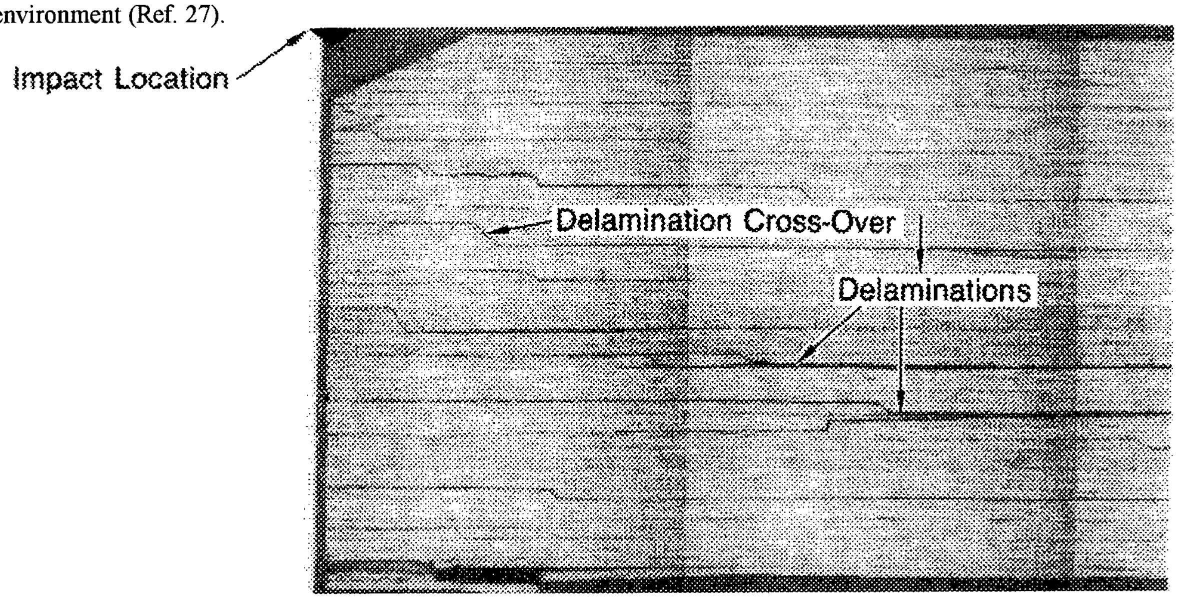

Figure 12. Impact Damage in Composites

Experimental data have shown (Figure 13) that impact damage can cause significant loss in strength. The degradation in compression strength is more severe than tension strength due to the delaminations between the plies caused by the impact damage.

Post impact fatigue behavior of a 42-ply composite laminate, subjected to sharp and blunt impactors, is shown in Figure 14. The fatigue testing was done at R=-1 (R being the ratio of maximum to minimum stress, R=-1 implies fully reversed tension compression). The fatigue responses of both types of impact are controlled by the static strength reduction and fatigue life-strain plots are flat.

2.2.1 REPAIR OF COMPOSITE MATERIALS

Repairs of composite materials are similar to those for metallic materials if mechanically fastened repairs are to be used. However, the repairs of composite materials are different from those of metals if the repairs are to be bonded. The process involved in making repair decisions is outlined in Figure 15. The damage must be evaluated and classified. If the damage is repairable, a decision has to be made whether to repair or replace a part. If the structure is to be repaired, additional decisions have to be made regarding maintenance level, where work will be done, kind of repair materials, and repair configuration. The first step in the repair of composite materials is to remove the damage area including the delaminated area in the impacted region. The next step is to clean the surface to be repaired and apply a bolted or bonded patch. These repair concepts are discussed in the following paragraphs.

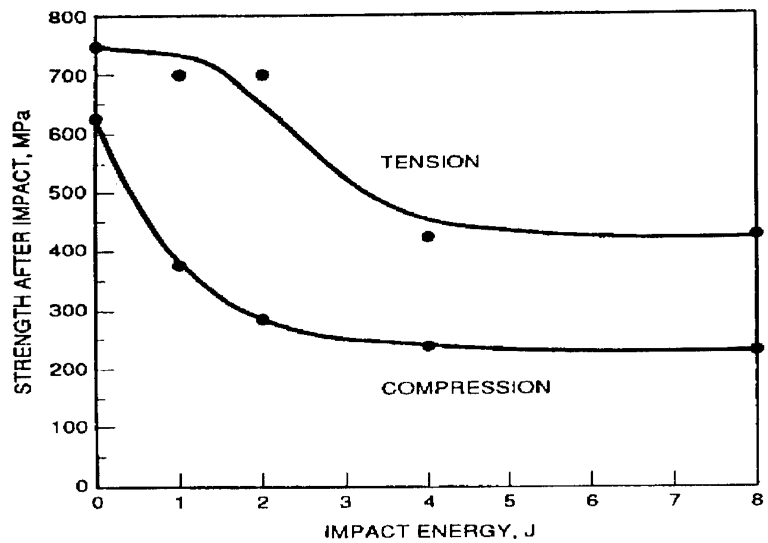

Figure 13. Strength Degradation Caused by Impact Damage

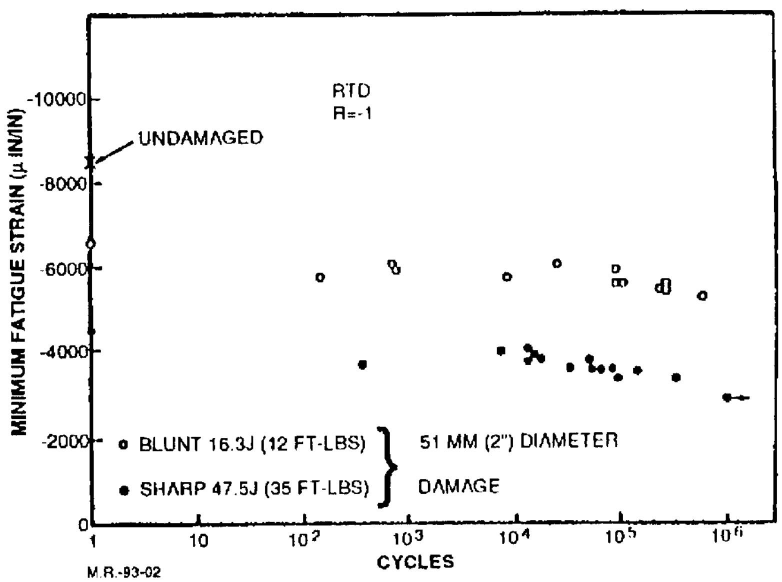

Figure 14. Fatigue Behavior (R=-1) of Impact Damage Composites

BOLTED REPAIRS

Bolted repairs for composite structures are similar to those for metallic structures. The major differences between the repairs for composites and metals are:

a) Different tools are used for drilling fastener hole in composites.

b) Special care is needed in drilling holes in composites to prevent splintering on the exit side of the hole. A back support is desirable.

c) Matrix in composite is brittle compared to metal, hence the fasteners that expand to fill the hole (e.g. driven rivets) are not suitable for composites.

d) Sharing of loads in different fasteners in composites is not uniform because composite materials do not yield as metals where the load distribution tends to be more uniform.

Three commonly used bolted repair concepts are shown in Figure 16 and are discussed here.

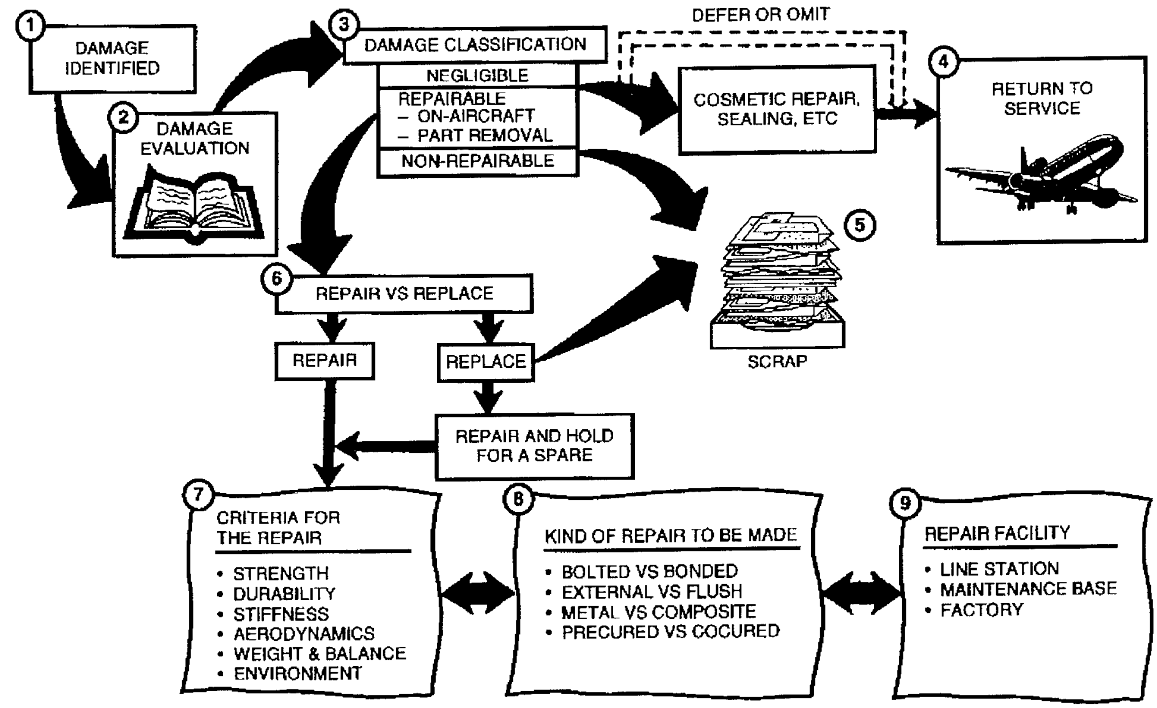

Figure 15. Damage Evaluation and Selection of Repair Methods

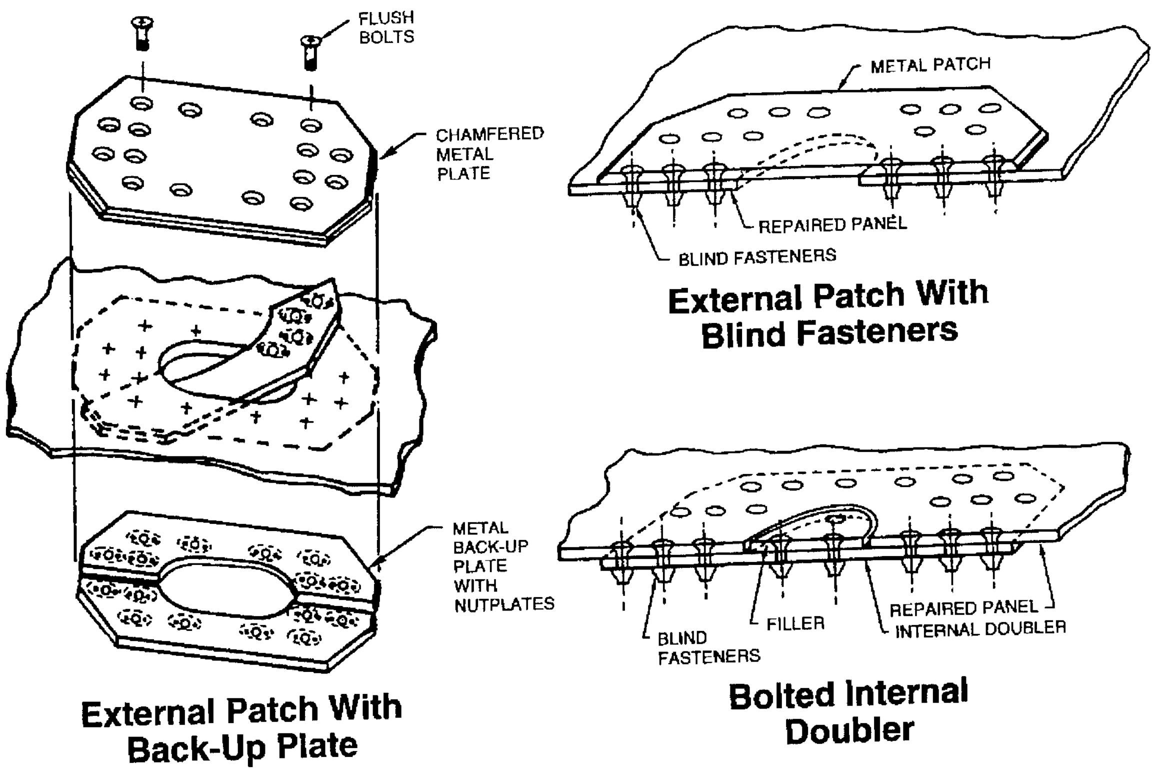

Figure 16. Bolted Repair Concepts

External Patch with Backup Plate- This concept uses an external chamfered metal patch bolted to the panel being repaired as shown in Figure 16. The bolts thread into nut plates mounted on metal backup plates that are on the side of the repaired panel. The backup plate can be split into two or more pieces and slipped through the opening as shown in the figure.

External Patch with Blind Fasteners-This concept is similar to the previous one, except that the backup plates are not used as shown in Figure 16. Blind fasteners are not as strong as bolts and nutplates, but if acceptable strength can be restored, this concept is easier to use.

Bolted Internal Doubler-This concept has been used as a standard repair for metal structures. Access to the back side is required to install the doubler as shown in Figure 16. The doubler cannot be installed through the hole as separate pieces because the doubler has to be continuous to carry loads in all directions. A filler is used to provide a flush outer surface, and is not designed to carry loads.

BONDED REPAIR CONCEPTS

Bonded repair concepts can restore greater strength to a damaged composite structure as compared to bolted repairs. External repair patches are suitable for thin skins, however, for thick skins the eccentricity of the external patch reduces its strength. Flush patches are preferred for thick structures, heavily loaded structures, or where aerodynamic smoothness is required. Commonly used repair concepts are step-lap and scarf repairs.

Step-Lap Repair- This repair concept is shown in Figure 17. The steps allow the load to be transferred between specific plies of the patch and parent material. This advantage tends to increase the strength of the joint, however, it is offset by the peaks that exist in the adhesive shear stress at the end of each step.

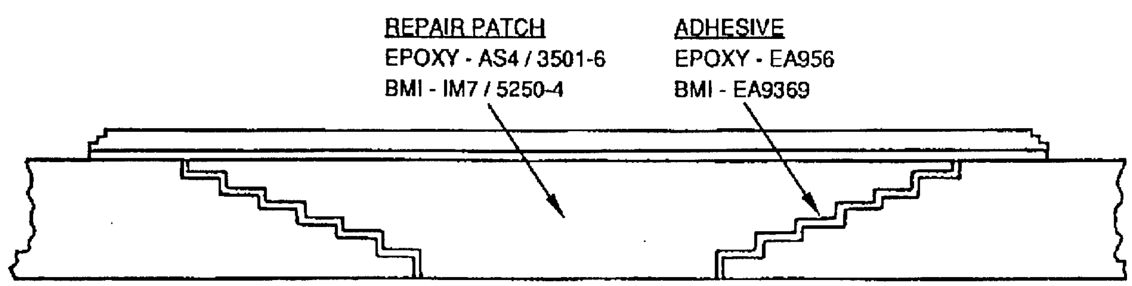

Figure 17. Step-Lap Repair

Scarf Repair- This repair concept is shown in Figure 18. The patch material is within the thickness to be repaired, with additional external plies added for strength. This configuration can restore more strength than an external patch as it avoids the eccentricity of the load path and provides smooth load transfer through gradually slopping scarf joint. A properly designed scarf joint can usually develop the full strength of an undamaged panel. The patch material is usually cured in place, and therefore must be supported during cure. While the patch material can be cured and then later bonded in place, it is generally difficult to get a good fit between the precured patch and the machined opening.

In practice, well made step-lap and scarf joints have approximately the same strength. A disadvantage of step-lap joints is the difficulty in machining the step to the depth of the exact ply that is desired on the surface of the step.

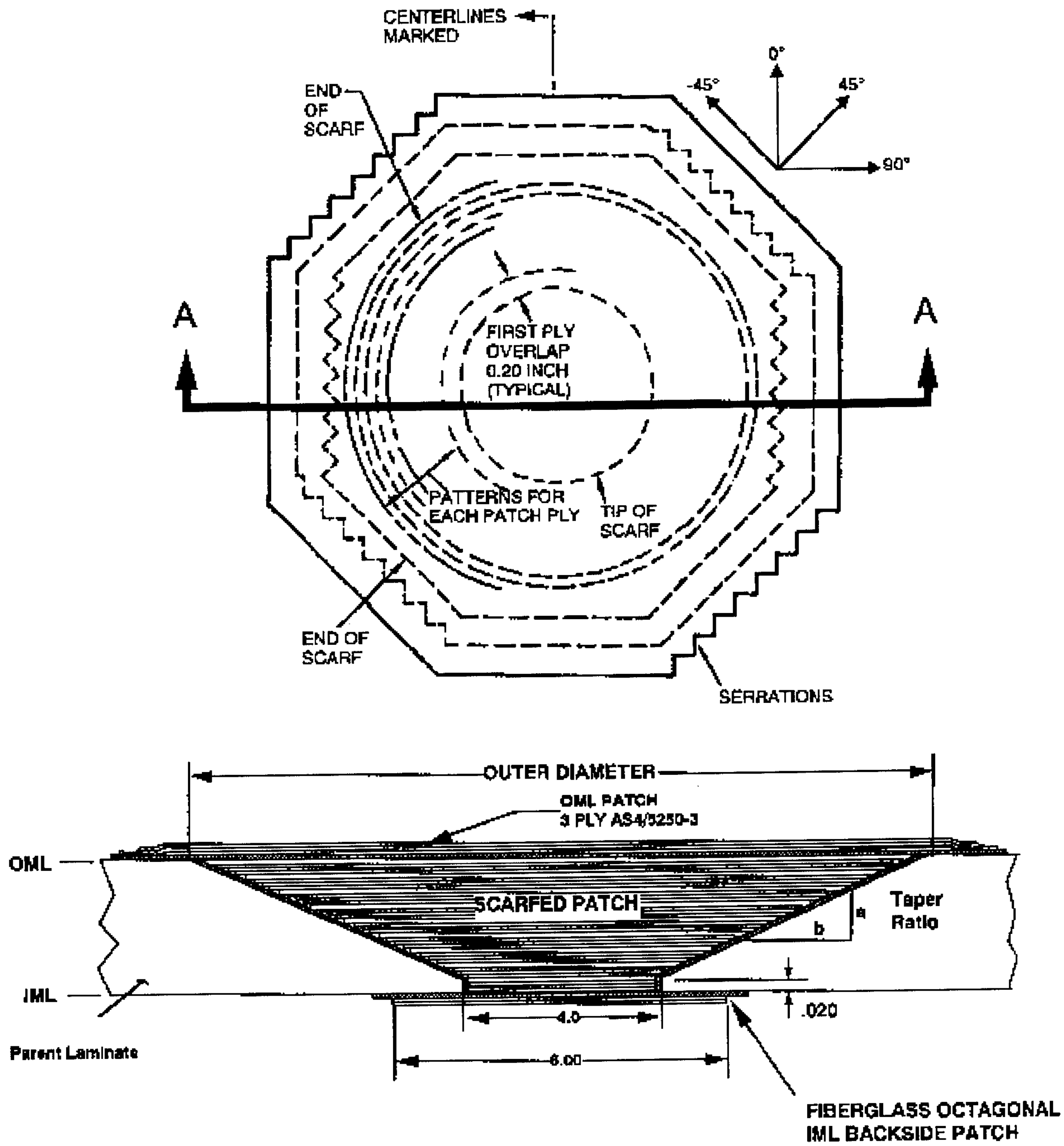

Figure 18. Scarfed Repair

2.2.2 Repair of Metallic Structures

Repair concepts for metallic structures are well established. The bolted repair concepts, discussed earlier for composites are applicable to metallic repairs. Improved structural efficiency of bonded composites has provided an excellent opportunity to repair metal structures with composites (Ref. 26, 28-35). In this repair concept a composite patch is bonded to the damaged metallic part instead of a conventional mechanically fastened patch. Bonded composite repair has many advantages over conventional mechanically fastened repair, namely: 1) More efficient load transfer from a cracked part to the composite patch due to the load transfer through the entire bonded area instead of discrete points as in the case of mechanically fastened repairs, 2) No additional stress concentrations and crack initiation sites due to drilling of holes as in the case of mechanically fastened repairs, 3) High durability under cyclic loading, 4) High directional stiffness in loading direction resulting in thinner patches, and 5) Curved surfaces and complex geometries easily repairable by curing patches in place or prestaging patches. The cross-section of a typical 16-ply graphite/epoxy patch bonded to an aluminum sheet is shown in Figure 19.

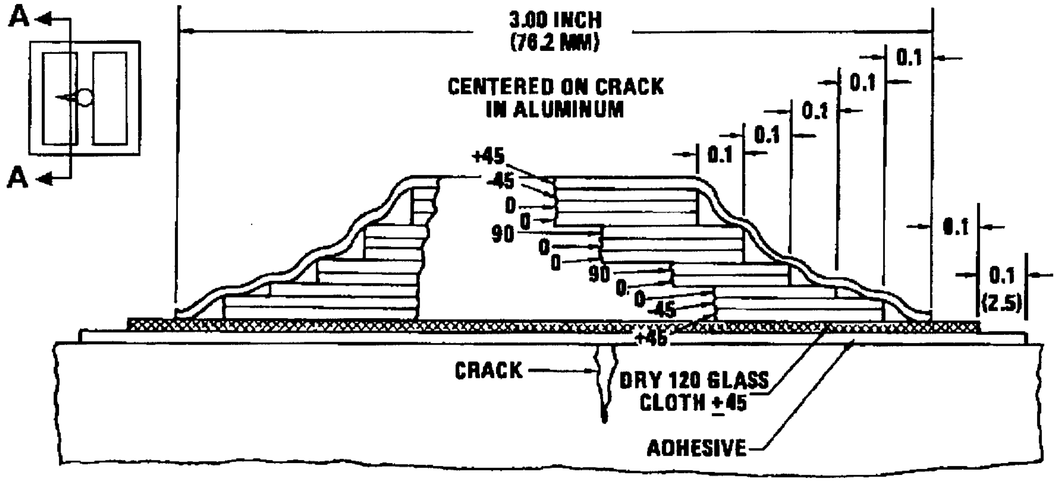

Figure 19. Cross-Section of a Typical Composite Repair Patch

The critical parameters for this type of repair are: 1) Surface preparation, 2) Adhesive material, 3) Composite repair material, and 4) Bonding operation.

SURFACE PREPARATION

Proper surface preparation is one of the most important considerations in bonded structures. The surface preparation process consists of paint removal, anodizing and priming. Liquid chemical paint strippers are not recommended as they may become entrapped in cracked areas and faying surfaces of adjoining structures, thereby causing a corrosion problem. Aluminum oxide abrasive cloth has been found to be suitable for small repair areas.

Both silane and phosphoric acid non-tank anodize (PANTA) have been found to be suitable. The silane process has the advantage of being non-acid process. However, from the point of view of long term durability of repair, the PANTA process may be desirable as sufficient test data is available on this process.

Primer is applied to the aluminum surface after anodizing with PANTA to prevent contamination and improve long term durability. BP-127 primer has been found to be suitable for FM-73 adhesive.

ADHESIVE MATERIAL

Room temperature cure adhesives are not considered suitable due to service temperature requirements of 180F (82C) in the majority of aircraft repair applications. Also, room temperature cure adhesives are paste adhesives and generally do not result in uniform bond line thickness in the repair, thus, affecting the load transfer to composite patch. Hence, high temperature film adhesives are preferred. Also, long term durability of room temperature adhesives is not well characterized.

A 350F (177C) cure film adhesive is not considered desirable as the curing at such a high temperature is likely to cause undesirable high thermal stresses. Also, an aluminum structure exposed to a 350F (177C) temperature will undergo degradation in mechanical properties. A 250F (121C) cure adhesive system is considered suitable for the composite patch repair of aluminum structure. Ductile adhesives such as FM-73 are preferred over brittle adhesives such as FM-400 due to the tendency of the brittle adhesives to disbond around the damage area, thereby reducing the load transfer to the repair patch.

COMPOSITE REPAIR MATERIAL

Both boron/epoxy and graphite/epoxy composites are suitable for the repairs. The choice between boron or graphite fibers should be based on availability, handling, processing and the thickness of the material to be repaired. Boron has higher modulus than graphite and would result in thin repair patches. Thin patches are more efficient in taking load from damaged parts as compared to thick patches. For repairing relatively thick parts, boron may be preferred over graphite.

It is considered desirable to use highly orthotropic patches, having high stiffness in the direction normal to the crack, but with some fibers in directions at 45 and 90 degrees to the primary direction to prevent matrix cracking under biaxial loading and inplane shear loads which exist for typical applications. This patch configuration can be best obtained with unidirectional tape. Woven material has greater formability and could also be used, although it would not make a very efficient patch.

The composite patches may be precured, prestaged or cured in place. For locations where vacuum bagging represents a problem, a precured patch may be prepared in an autoclave and then secondary bonded to the repair area. For relatively minor contours, a prestaged patch may be used. For curved surfaces the patch may be cured in place during the bonding operation.

BONDING OPERATION

Bonding of repair patches requires a proper temperature control within +10F and -5F in the repair area. Thermal blankets are available to provide temperature in excess of 1000F (538C). A proper temperature control within tolerances is necessary for bondline to achieve desirable strength. A large aircraft structure compared to a small repair area may act as a heat sink and jeopardize maintaining desired temperature control for the required duration. Proper heat blankets for surrounding areas may be required for such cases.

LIFE EXTENSION WITH COMPOSITE REPAIR PATCHES

The crack growth data obtained from a repaired center-crack panel (7075-T6 aluminum, 0.063 inch (1.6 mm) thickness) are shown in Figure 20. It is seen that starting with the same initial crack length, the panel without a repair patch fails after about 870 missions (0.92 life time) at a crack length of 1.36 inch (34.6 mm). The panel with the repair patch did not fail even after 2350 missions (2.5 life times) at a crack length of 1.93 inches (49 mm). Thus, a considerable extension in life was obtained with the composite repair patch.

COMPARISON OF ANALYTICAL AND EXPERIMENTAL RESULTS

The crack growth behavior of the cracked panel with a composite patch was predicted using analytical stress intensity factors (Ref. 35-36) for the patched structure and the crack growth data, obtained on an unpatched center crack specimen. Comparison of observed and predicted fatigue crack growth behavior in a 7075-T6 aluminum 0.063 inch (1.6 mm) thickness repaired with a 3 inch (76 mm) square 12 ply graphite/epoxy patch, moisture conditioned to one percent moisture, is shown in Figure 21. It is seen that the correlation between predicted and observed crack growth is excellent. The specimen did not fail even after two life times of spectrum loading.

BORON VERSUS GRAPHITE REPAIR PATCH

Two identical specimen were tested (Ref. 29) under spectrum loading- one repaired with 12-ply graphite/epoxy patch and the other with 8-ply boron/epoxy patch. Both the repair patches had identical Et (modulus in loading direction times thickness). The comparison of crack growth in two specimen is shown in Figure 22. The figure shows identical crack growth in the two specimens. This indicates that the load transferred to 12-ply graphite/epoxy and 8-ply boron/epoxy patches is identical. This is predicted from analysis as both patches have identical Et (Ref. 35-36).

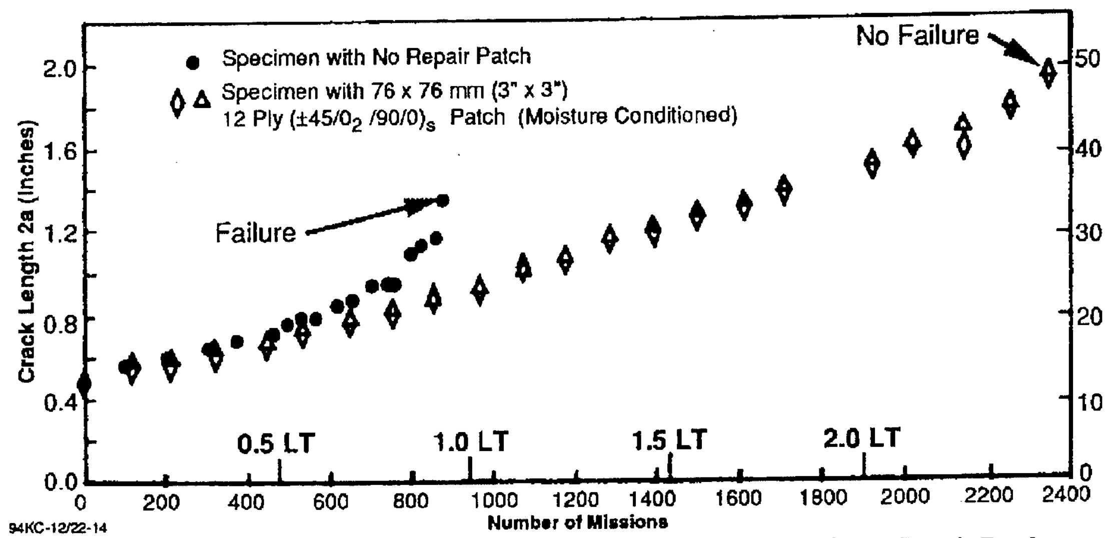

Figure 20. Comparison of Crack Growth in Specimen With and Without Repair Patch

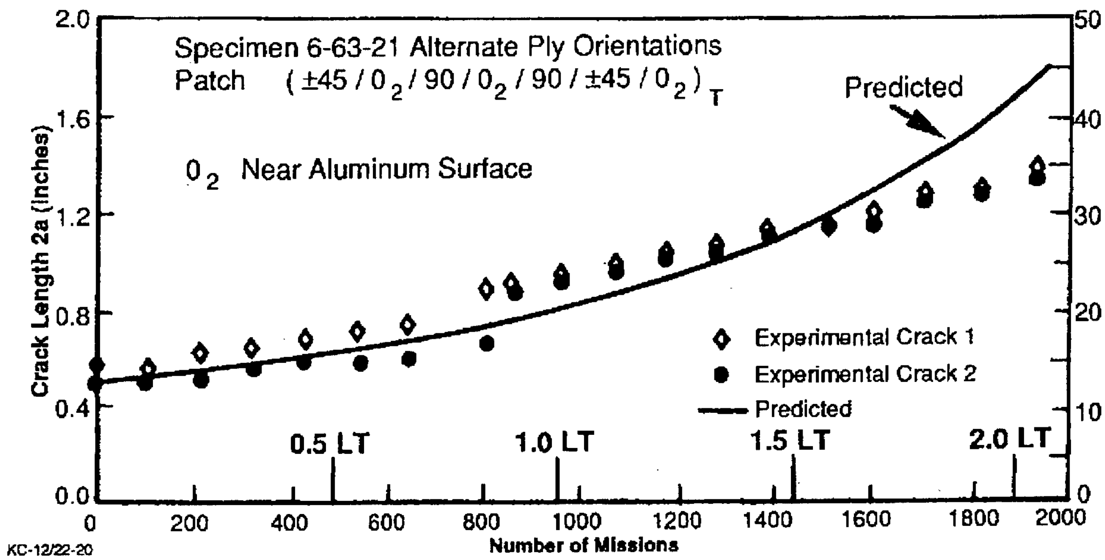

Figure 21. Comparison of Observed and Predicted Crack Growth

REPAIR DESIGN FOR NO DAMAGE GROWTH

It is possible to design composite repair patches so that the damage in the repaired structure will not grow. Of course, the feasibility of such a design depends on the stress level, the type of material to be repaired, material thickness, the crack length to be repaired and spectrum. In the majority of transport aircraft where design stress levels are relatively low, it is possible to design repairs such that the damage does not grow. This is particularly true for fuselage structures where material is predominantly 2024-T3 aluminum and gauge thicknesses are small. Crack growth behavior in 2024-T3 material 0.032 inch (0.8 mm) thick specimen, repaired with 12-ply Gr/Ep patch is shown in Figure 23. No crack growth in two lifetimes of spectrum loading is seen. Thus, the repairs can be designed for no damage growth and there by eliminating inspection requirements.

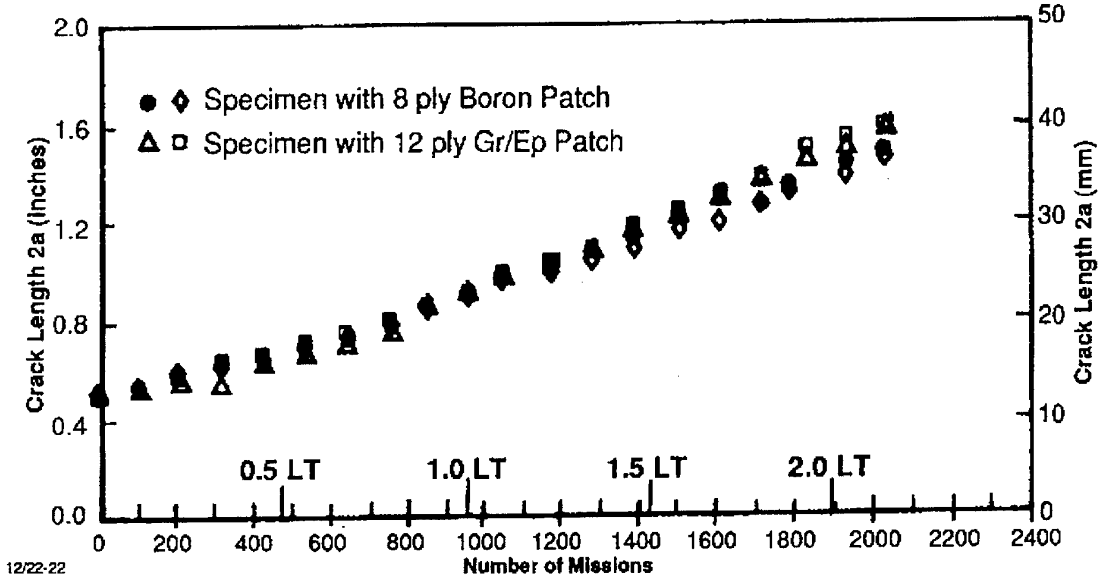

Figure 22. Crack Growth in Specimens With 12-ply Gr/Ep and 8-Ply Br/Ep Patches

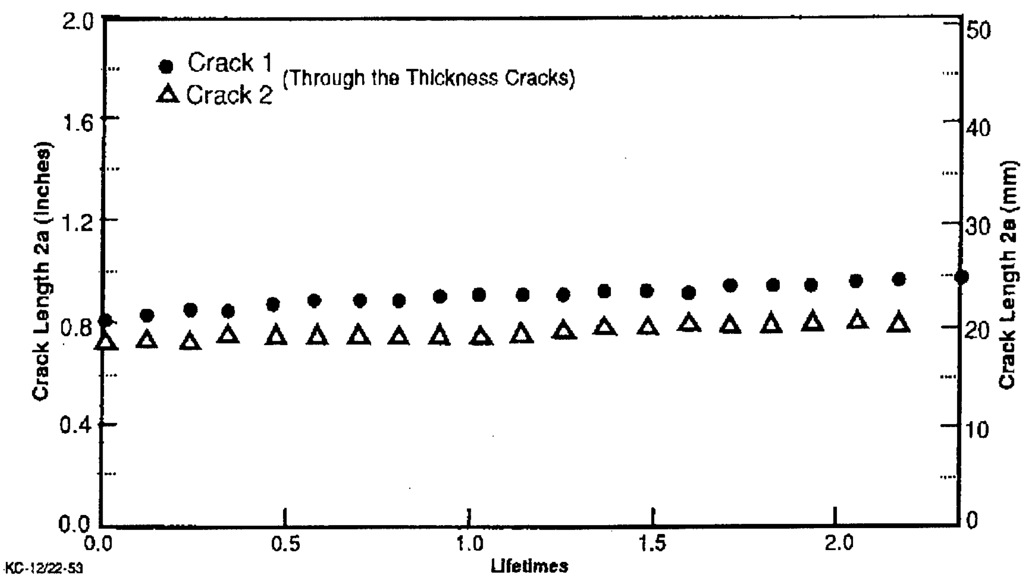

Figure 23. Fatigue Crack Growth in 2024-T3 Aluminum, 0.032 inch (0.8 mm) Thick With 12-Ply Gr/Ep Patch

REPAIR OF CRACKS AT LARGE HOLES

One specimen having a 0.1 inch (2.5 mm) crack emanating from a 1.0 inch (25 mm) diameter hole was repaired with a graphite/epoxy patch on the cracked side with back to back patches on both faces (Ref. 28) as shown in Figure 24a. The specimen failed after 1.41 lifetimes of spectrum loading due to the initiation of a crack on the unpatched side. Another identical specimen was repaired with patches on the cracked side as well as the uncracked side of the hole on both faces as shown in Figure. 24b. This specimen did not fail even in five lifetimes of spectrum load testing. This result indicates that one needs to be careful in the design of repair patches to make sure that the load redistribution does not create problem elsewhere. Also, these results show that the repair patches may be used to reduce stress concentration and there by increase fatigue life. If in-service experience shows crack initiation at certain locations in structures, it is desirable to reinforce these areas by bonding composite patches to increase the life of the components and reduce inspection requirements.

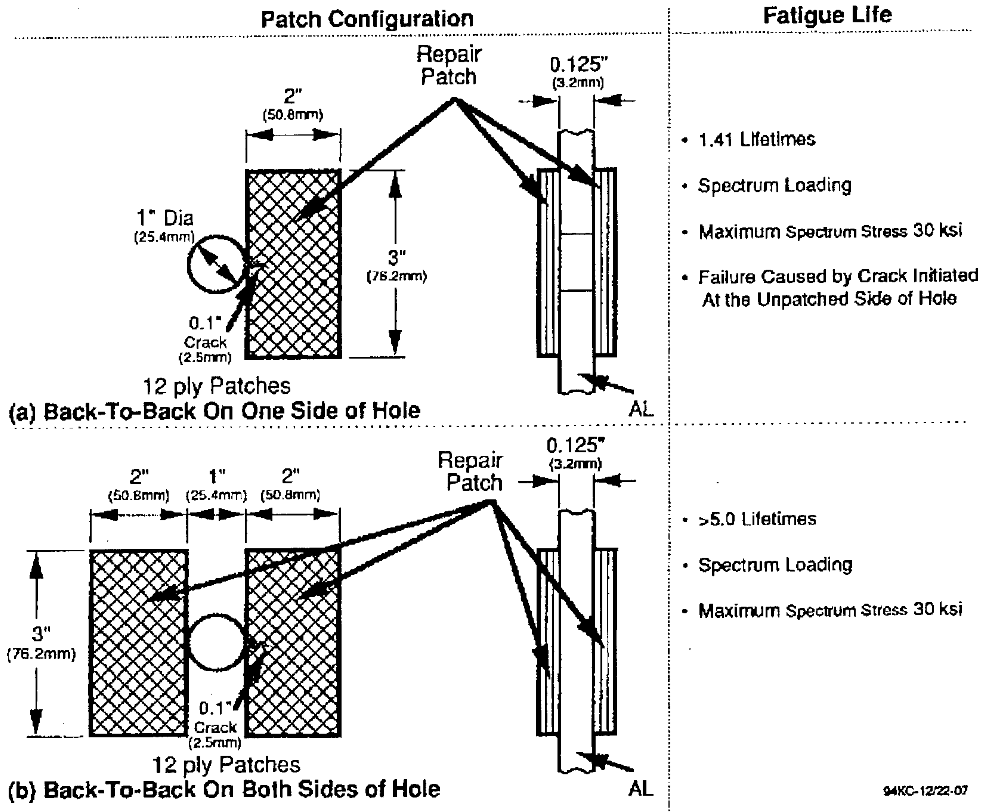

Figure 24. Composite Patch Repair at Large Open Hole

3. ADVANCED MATERIALS APPLICATIONS

The majority of the maintenance problems in the aging aircraft fleet are due to the use of alloys such as 7075-T6 and 7079-T6 having poor stress corrosion cracking (SCC) resistance. To reduce maintenance cost, these alloys should be replaced with materials that have high strength, good SCC and improved durability and damage tolerance. Some of the alloys/tempers that were used for older aircraft would no longer be used today. For example, 7075-T6 and 7079-T6 would not be selected for stress corrosion-prone applications because materials such as 7075-T73 and -T76 have been developed for use where improved corrosion resistance is needed. A general practice is to replace T-6 with T-7 alloy. The substitution has not been easy as T-7 has lower mechanical properties (ultimate and yield) compared to the T-6 alloy as shown in Figure 25. Hence the design of parts with T-7 material needs to be beefed up (increased thickness), resulting in weight penalty. Also, in some cases the thickness cannot be increased due to the restriction on dimensions dictated by the presence of mating surfaces. New materials such as the Powder Metallurgy (PM) alloy 7093-T73 can off-set the disadvantages of 7075-T7 aluminum as it has mechanical properties which are better than those of 7075-T6 as shown in Figure 25. Any redesign of structural components with PM alloy has no restrictions and will result in weight savings.

The atmospheric corrosion resistance of conventional and PM alloys is shown in Figure 26. The figure shows PM alloy to have atmospheric corrosion resistance far superior to that of T-6 and some what better than T-7. The notched fatigue behavior of PM alloy 7093-T7 is shown in Figure 27 for stress concentration factor Kt = 3.0. The figure shows PM alloy having far superior fatigue life compared to other aluminum alloys. The improved properties of 7093-PM alloy indicate that the spare/retrofit parts, designed with the alloy, will result in enhanced durability and damage tolerance. In addition, weight savings, increased service life and lower support requirement can be achieved with the alloy.

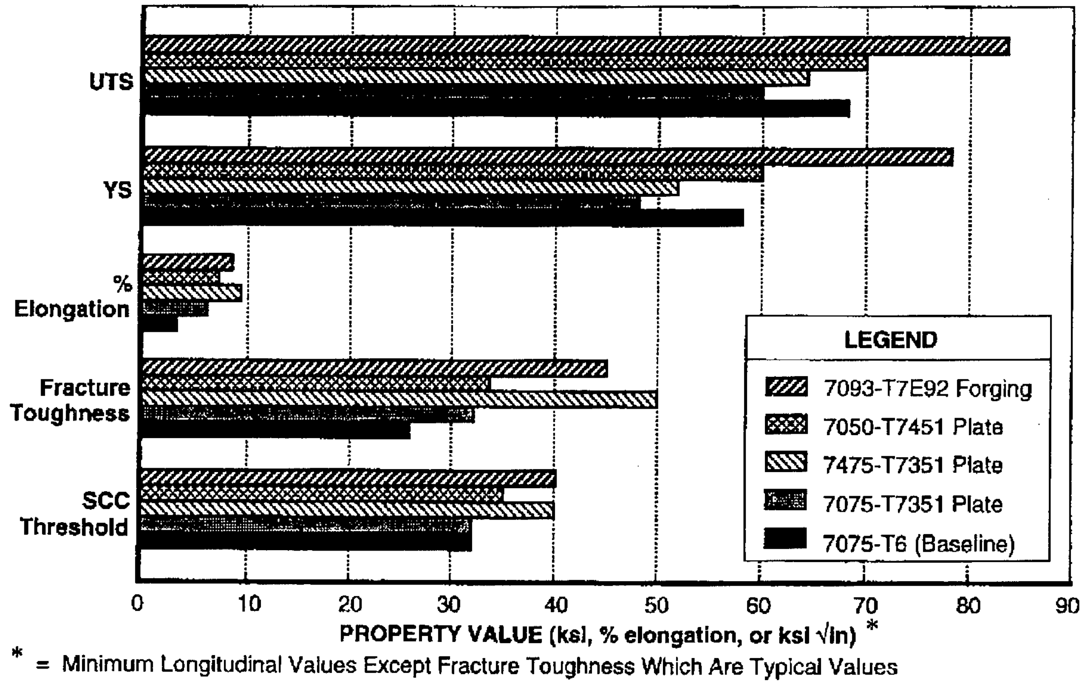

Figure 25. Mechanical Properties of Typical Aluminum Alloys

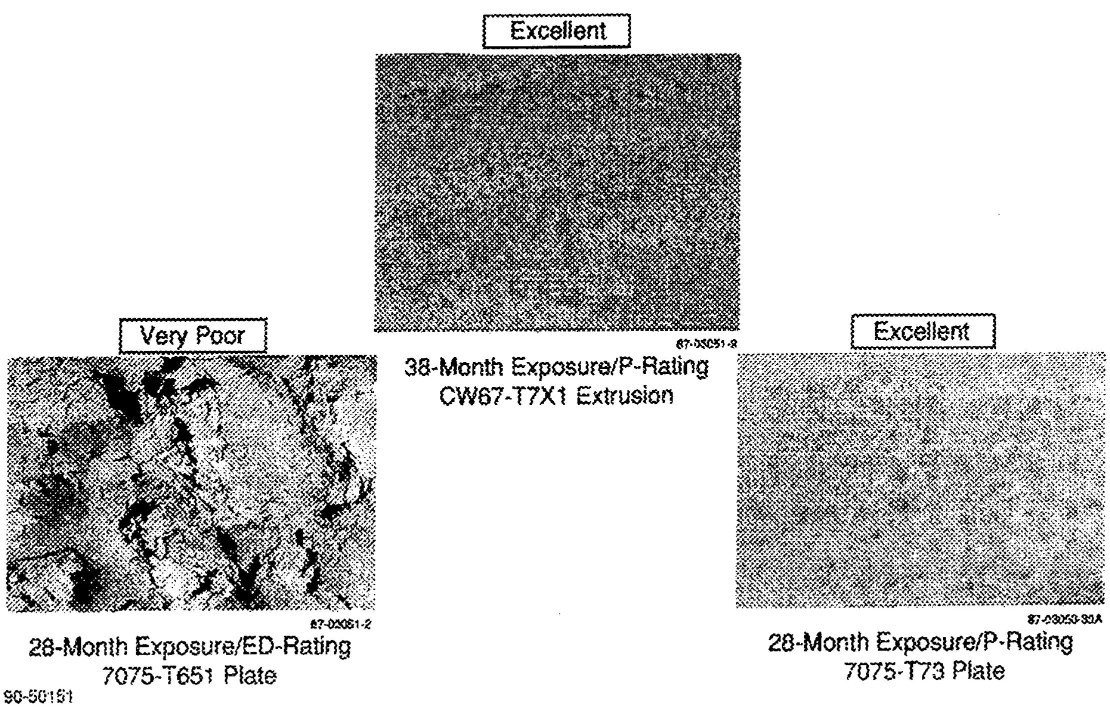

Figure 26. Atmospheric Corrosion Resistance of Various Aluminum Alloys

In-Service Evaluation of 7093-T7 PM Aluminum Alloy

T-38 parts with structural problems were considered for an in-service evaluation of 7093-T7 PM aluminum alloy (Ref. 37). Since peak aged 7075-T6 and 7079-T6 aluminum were the standard high strength alloys used in the design of the aircraft for thick product forms, and these alloys are highly susceptible to SCC, the components from this aircraft proved to be the prime candidates for replacement with 7093 PM alloy. Components were chosen based on severity of the problem, flight criticality, fatigue requirements, size of the part and ease of the replacement. Candidate parts were-1) Outboard engine mount support, 2) Jack pad support fitting, and 3) Speed break support beam.

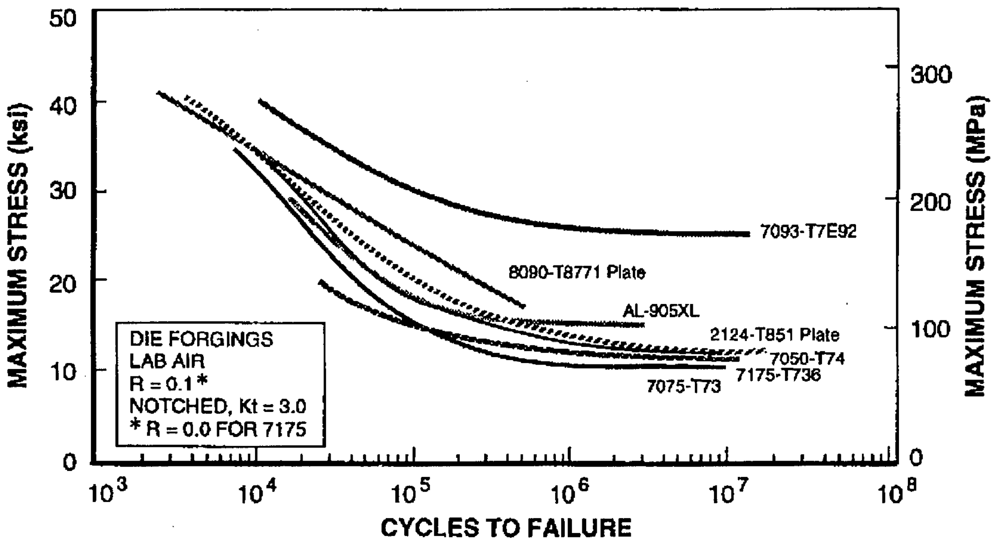

Figure 27. Notched Fatigue Life of Various Aluminum Alloy Forging

All three candidate parts provided a number of benefits for choosing any of the parts. However, outboard engine mount support (Figure 28) was selected as the best overall part for solving an existing problem while gaining in-service experience. Mechanical properties, obtained from ten forging were found to be within acceptable scatter band. A few engine mount supports have been flying without any known in-service problems.

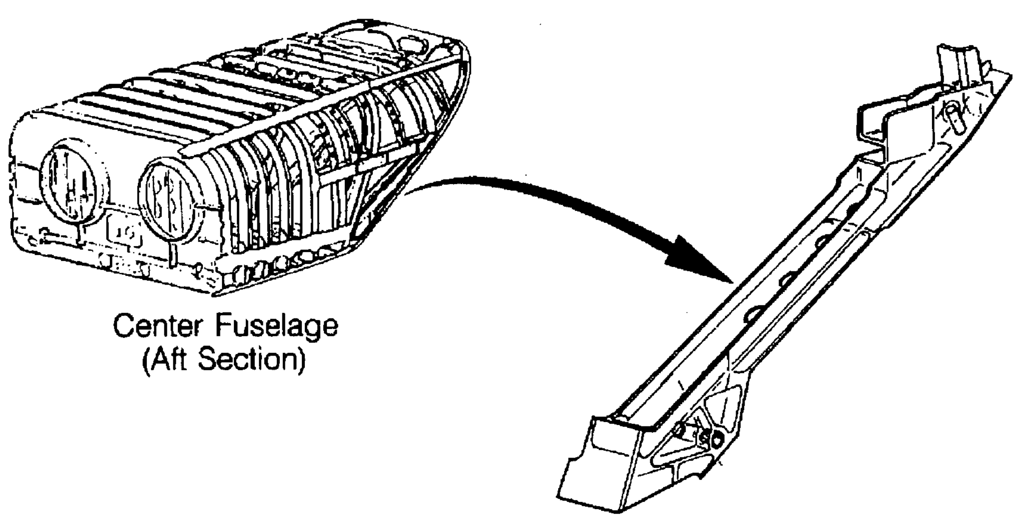

Figure 28. Engine Mount Support

4.0 CONCLUDING REMARKS

The life enhancement technologies and advanced materials have provided excellent opportunities to fulfill aging aircraft needs such as:
1) Reduced life cycle costs
2) Reduced/eliminated repairs
3) Reduced/eliminated inspections
4) Simplified maintenance
5) Reduced support requirements
6) Fulfilled severe usage requirements
7) Extended airframe life
8) Improved payload
9) Reduced structural weight

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Aircraft Loads and Monitoring

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SUMMARY
The life of a weapon system is influenced to a high degree by the structural integrity of the airframe. Numerous programs to ensure this have been established within NATO´s Air Forces. Structural loads, leading to fatigue as well as corrosion, depending on the usage environment, are the major reason for degradation of structures. The many different classes of loads, the generation of loading conditions during the design phase, consideration of static and fatigue loads for structural lay-out and their validation are presented.
The fatigue life of aircrafts in service is different from the design life for many aircrafts not only due to the extended need for the airframe as a platform for new/upgraded systems (life extension), but also due to different usage compared to the design spectrum. Monitoring of the life consumption is therefore essential. Methods and concepts to establish the "used life" are described for two different aircrafts and the influence of A/C-roles and -equipment as well as structural weight increase over time are discussed.

0. BACKGROUND

The effectiveness of military force depends in part on the operational readiness of aircraft which itself is largely dependent on the condition of the airframe structure. This condition again is affected by a number of factors among those the physical loads in various forms together with the used life of the airframe are important. With increased and extented usage of airframes in all airforce inventorie and the requirement for various role changes the subject of airframe loads assessment, -qualification and aircraft loads-monitoring becomes more important, not only for flight safety but also and with an increasing tendency for economic reasons.
A general understanding of the various types of airframe loads, their generation and application during the design process, the transfer processes from "external loads" into "structural loads", loads qualification during ground and flight testing is therefore of equal importance to the wellknown process of usage monitoring and derivation of usage factors from the different fatigue tests or the set-up of structural inspection programs.
When structural life of aircrafts are discussed, often the flight hours or number of flights are still considered the governing factor, sometimes adapted with factors on "damage hours" or "usage", while from a structural engineering viewpoint the operational stress spectrum and therefore the life on the different aircraft components are not only a matter of flight hours and spectrum ratio but also driven by modification status, structural weight status and role equipment.
This paper describes loads- and loads monitoring-activities during the three major phases of the life of an airframe, where structural loads and their influences on the airframe condition are vital to the structural integrity and the economic usage of the weapon system:

* The structural loads during design and Qualification of A/C structures
* Loads Monitoring and "Fatigue Life" of airframes
* Impacts due to A/C modification and Role changes.

Trends with respect to the increased usage of therotical modelling and extended databases for flight parameter recording of individual aircraft are also discussed.

1. STRUCTURAL LOADS DURING THE DESIGN AND QUALIFICATION OF A/C STRUCTURES

Loads are accompanying an aircraft's life from "the cradle to the grave". Each change on the A/C in principle influences the loads. Fig. 1-1 gives an idea how loads are generated and what they are good for.. .

1.1 Loads and Fatigue

Aging aircraft does not only mean that an aircraft is getting older, it also means that basic design criteria have changed during time:

- mass growth
- enhancemant of performance, e.g. engine
- new configurations (stores)
- update of FCS
- mission profiles
- additional/changed roles
- actual usage spectrum
- etc.

Most of these changes have an immediate impact on aircraft loads, others will not change load levels but may change the underlying statistics, e.g. fatigue spectra.

The determination of loads for all important structural components is a main prerequisite for correct and successful design of an aircraft together with the

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qualification for strength and fatigue by calculation and test. Exaggerated one could state that without loads no structure would be needed regarding strength.

Whereas for transport aircraft with their rather limited range of operational manoeuvers fatigue is a main design driver with respect to structure, fighter aircrafts are predominantly designed to (static) limit load cases, which in general cover a lot of strength required for fatigue.

But this is only true as long as fighter life does not exceed the originally planned lifetime and the roles, missions etc. are compatible with the design criteria at the beginning. Therefore in the case of aging aircraft fatigue is an ever more important issue, which may be highlighted and assessed by considering loads and loads spectra.

Admittedly in many cases there is no simple one to one relationship between "external" loads and local internal stress, which after all is the basis for the assessment of lifetime consumption resp. remaining life for structure components. But providing loads are known for a special structural interface or component, reliable conclusions can be drawn regarding local stress relating to the manifold of load cases from experience, measurement and detailed FE analysis during design and test phases.

In addition the comparison of load spectra may be suitable for drawing conclusions without recourse to stress calculations.

The notion of loads being such a central point for the understanding of the aircraft design process and especially the context with fatigue life time it will be shown in the following paragraphs how

- Design loads are assessed
- External loads are converted into structural airframe loads
- Aircraft structural loads are qualified by calculation and test (e.g. windtunnel tests - pressure plotting model, flight test - flight load survey, etc.)

In addition it is envisioned, that from pure calculation of loads, backed up by a (statistically) safe correlation of manouvre loads and local stresses, a survey of fleet and individual aircraft lifetime can be achieved, realizing a concept which is highly illustrative.

1.2 The Determination of Design Loads

In the following a summary is given on the methods how design load cases are determined, special attention is paid to points where an immediate context with fatigue exists. Fig. 1.2-1 shows the "loads loop" which usually is repeated several times in the different stages of the A/C design. First of all the *Structural Design Criteria (SDC)* are prepared as a basis for design, then a *Loads Model (LM)* is built, based on the SDC, and a manifold of input data. The LM includes a collection of program modules, which guarantee that the load cases selected for design are provided to Finite Element (FE) analysis in the form of an overall balanced distribution of loads on the FE grid nodal points. Thus, starting with the SDC the load loop ends with the preparation of nodal point loads for stress analysis.

Usually an improved or changed data basis results in an update of the LM and consequently in more accurate and more detailed design load cases. Typical improvements are a better aerodynamic data basis or a refined FE-model. Modifications in the mass status, control laws etc. may result in substantial changes of loads.

It must not be emphasized how important - especially for fatigue - an exact knowledge of structural stress distributions is and therefore of external loads. Computers play an important part with resp. to better results in the assessment of loads. Whereas in the past the available computer power was rather poor and strong software tools were scarce goods, today there are virtually no limits, from this side.

Most of the ageing A/C fleets of the NATO airforces are designed and flight tested by the end of the sixties respectve the beginning seventies, e.g. Tornado, Harrier, F-16, F-18, Mirage 2000 etc. An A/C like the Phantom even emerged already in the fifties.

Compared to fighter A/C designed during those past decades it should be pointed out that in the meantime the conditions for A/C design have extremely changed, in detail:

- much better tools, soft- and hardware, and with that a very intensive investigation to calculate and control limit and fatigue loads (substantial increase in the number of component load monitoring stations)

	Tornado IDS	EF 2000
Basic Load Cases (BLC) Flight and Ground HandlingLoads	33	105
Unit Load Cases (ULC) Hammershock, Engine Thrust, Airbrake etc.	12	16
Combined Load Cases Superposition of scaled ULCs to BLCs	≈ 100	590

Number of Load cases for FEM Analysis (Check Stress)

- more accurate databases
  - because of Carefree Handling Flight Control Systems (FCS)
  - A/C mass distributions
  - A/C aerodynamics calculated with mature CFD (Computational Fluid Dynamics) methods and measured much more reliable in the wind tunnel or in flight tests.

- coupling of structure and aerodynamics (Aeroelastic effects) available from the beginning
- FE models
- Loads Model

• extensive flight testing, especially *flight load survey*
• more structural ground tests
• reduced ultimate factor (1.4 instead of 1.5), among other things a consequence of modern FCS and accurate knowledge of loads

That means that the static design of "old" A/C usually is rather on the safe side. With respect to fatigue the situation is not so good because lacking a powerful tools like a balanced LM, one procedure was balancing loads over the A/C in an artificial way in those days, and therefore parts of the structure not immediately under survey got loads which were calculated on rough assumptions only.

1.2.1 Structural Design Criteria (SDC)

Aircraft Loads are determined according to requirements and regulations collected in a document called *Structural Design Criteria*, i.e. a system specification with respect to loads and structure.

Some of the more important items are:
Design masses are defined for different flight conditions to cover the whole mass and center of gravity (C.G.) range, e.g.:

- basic flight design mass
- landing design mass
- maximum take off mass
- etc.

Total mass and mass *distribution* not only affect loads on wing as is sometimes believed but loads on most parts of the aircraft structure. Design mass is one of the most important criteria for structural design. For example the basic flight design mass is coupled to the max/min allowed Nz, for higher masses the rule:

$$Nz \cdot Weight = const$$

applies.

V-n Diagrams

define the regime of speeds in combination with max/min allowable *load factor Nz* including gust conditions, see Fig. 1.2.1-1. For low speed the limit Nz depends on max lift and dynamic pressure whereas for higher speed Nz is limited by the structural strength of the A/C.

Flight Envelope(s)

This defines the operating range with resp. to Mach-Altitude, for which the A/C is designed. Limits are determined by attainable Nz, temperature etc. Fig. 1.2.1-2 shows flight envelopes for the Tornado A/C. For an fixed wing A/C usually only one flight envelope diagram has to be defined, but an A/C like Tornado presents an additional complication as each (fixed) sweep position has to be considered as a different A/C. This is clearly seen by the different flight envelopes for the shown sweep positions.

Fig. 1.2.1-3 indicates what part of the flight envelope is of importance for the investigation of loads and shows points in the Mach-Altitude range for which loads are calculated according to the scheme explained later. The points are selected to cover all essential effects due to high Nz, incidence, roll rate, gust, Mach effects etc. The (nonlinear) effect of flexible aerodynamics is the main reason that so many "interior" points in the Mach-Altitude range ("points in the sky") are of importance.

Environmental Conditions define

System pressures
Cabin pressures
Temperatures
Local accelerations for qualification of equipment

Performance Requirements (Aeroelastic effectiveness etc.) are to be fulfilled

Example: Due to aeroelastic deformation under load the effectiveness of a control surface may be reduced substantially, for differental tail planes even roll reversal may occur. Therefore a specification by the customer may be the max. allowable degradation in efficiency under such circumstances. This means that an optimization of flap structure and tailoring must be carried out to ensure a required roll rate.

Former practice was rather to find out effectiveness after design had been completed.

Configuration specification (External Stores, Control Surface Schedules etc.)

Store configuration definitions can have great impact on fatigue due to load alleviation by inertia effects (stores on wing e.g.). See also fig. 1.3-2

Fatigue Load Spectra are usually agreed on with the customer. A discussion of this point can be found in chapter 3. An important definition is the scatterfactor to be applied for tests.

Many of the SDC requirements come from the customer, others are prepared in cooperation between customer and contractor. The SDC are subject to revisions also during the design process.

1.2.2 Aircraft Loads

The quality of loads acting an an A/C are of a different kind. The following grouping shall give an idea of the loads to be considered during design:

Quasi-static loads:

Flight Loads:
- Symmetric manoeuvres
- Asymmetric manoeuvres
- Deep and flat spin
- Gust loads

Ground Handling:
- Take off
- Landing
- Repaired runway
- Taxiing (asymmetric braking, turning etc.)
- Towing, Pivoting etc.

Local and Internal Loads:
- Max./min. aerodynamic pressures (skin)
- Local accelerations
- System pressures
- Bay pressures (pressurized areas)
- Hydrostatic pressures (tanks)
- Intake duct pressures (steady state and hammershock conditions)
- Engine thrust

Dynamic Loads:

- Buffet (Wing, Fin Buffet etc.)
  Flight measured buffet on the fin is shown on Fig. 1.2.2-1, clearly demonstrating the importance of buffet for fatigue if high angles of attack are flown.
- Dynamic Gust
- Vibrations
- Acoustic Noise
- Flutter
- Shimmy (Undercarriage)

Fatigue Loads:

Fatigue load cases are derived from the static and dynamic load conditions if applicable, e.g. if the frequency of resp. load cycles is sufficiently high (hammer-shock will certainly not be a fatigue case).

The above static, dynamic and fatigue loads have to be combined with the corresponding temperatures, i.e. cold day, hot day, and moisture conditions especially for new materials, e.g. CFC.

1.2.3 Flight Parameter Envelopes

Loads are not a function of Nz alone but depend substantially on many other flight parameters, the most important of which are

Incidence
Sideslip (for design the significant factor is $\beta \cdot Q$, the product of sideslip and dynamic pressure)
Control deflection angles (aileron, rudder etc.)
Lateral load factor Ny
Vertical load factor Nz
Roll rate
Roll acceleration
Pitch acceleration
Yaw acceleration

Usually less important:

Longitudinal load factor Nx
Pitch rate
Yaw rate

Adequate combinations of those parameters - as occuring during real flight manoeuvres - can yield high loads on different parts of the aircraft structure, even for rather moderate vertical load factors. In order to illustrate this context, Fig. 1.2.3-1 shows flight parameters during a typical MIL pitch manoeuvre and indicates that for certain time instants the force on the tailplane (=T/P) has a maximum dependant on the flap deflection, incidence angle, pitch acceleration and Nz.

Therefore it is the engineers skill to find all the critical combinations for the different A/C configurations and the possible manoeuvres in the whole flight regime. Regulations like Mil-Spec for fighter aircraft or FAR for other A/C provide a good guide to determine the critical combinations of flight parameters for design, at least in times of stable A/C and with conventional FCS. Very often it is desirable to determine flight parameter values from response calculations, using an A/C's response and loads simulation program.

However, in the early and mid stages of modern fighter A/C design a reliable model of an FCS usually is unavailable, therefore agreement between specialists of different disciplines (aerodynamics, flight mechanics, loads etc.) on *flight parameter limits* in the form of *envelopes* is the adequate means to cope with such difficulties. Fig. 1.2.3-2 shows typical envelopes as used in the first design phases with the envelope corners annotating the impact on loads for different A/C components.

1.2.4 The Loads Model

The LM is the central tool for running the "loads-business". It presents a model (on computer) of the total A/C, integrating the physics of motion, the underlying aerodynamics, SDC etc. and has interfaces to stress (FEM) and other disciplines, in detail:

The LM is a collection of all *input* data relevant for the calculation of (static) loads like
- Wind Tunnel and Flight Test aerodynamic data
- FEM-grid including stiffness matrix
- Masses and mass distributions
- FCS program module (for simulation of flight load specific manoeuvres and landing cases)
- Aerodynamic surface grid

and provides a *computer program* to determine loads and load-specific data like:

- Pressure distributions as a function of Mach number, incidence, control deflections etc.
- Calculation of aeroelastic effects from the coupling of structural flexibility and loads (aerodynamic and inertia)
- Aerodynamic derivatives for total A/C (used to simulate A/C motion) and A/C component aerodynamics, harmonized with resp. to flight test and wind tunnel data
- Manoeuvre response simulation and interface loads (at component monitor stations), calculation for preparation of component loads envelopes
- Landing gear model and landing simulations (flexible A/C) with structural loads calculations
- Generation of loads distributions along structure components
- Distribution of design loads on nodal points of the FEM for stress analysis

and makes available a (ever growing) *data base* of

- Flexible aerodynamics (A/C components and total A/C) for the whole Mach/Altitude range
- Manoeuvre response and load cases
- Nodal point distributions for design load cases

Of course the implementation of the LM is governed by the SDC.

One of the focal points realized by the LM is the fact, that all (design) load cases are calculated as *balanced* load cases, i.e. all conditions with resp. to aerodynamics, mass distribution and flight manoeuvre match perfectly and provide the correct loads for each structure item for any load case. In other words, the sum of net[1] forces and net moments over all points of the structure must be zero:

$$\Sigma_{x,y,z}\ F(x,y,z) \equiv 0 \quad \text{and} \quad \Sigma_{x,y,z}\ M(x,y,z) \equiv 0$$

As mentioned above, such a complete LM was not available for A/C like Tornado.

1.2.5 Aircraft Component Loads and Design Cases

Loads may be calculated in 3 degrees of refinement:

- Interface or *component* loads
- Load *distributions*, e.g. bending moment along wing span, usually one dimensional
- *Nodal point* loads (FEM)

The latter two are suitable to stress analysis and dimensioning and are usually only applied to design load cases. Component loads, however, are used to find the *design load cases*, which usually are different for different structure locations. Therefore the A/C structure is divided in components, with the boundaries representing main constructive items like interfaces, bulkheads etc.

An example can be seen on Fig. 1.2.5-1, showing the *A/C components*

- Wing
- Wing spoiler
- Front fuselage transport joint
- Fwd front fuselage
- Radom
- Rear fuselage transport joint
- Taileron
- Fin
- Rudder
- Airbrake

The resp. *load monitoring stations* are also shown in the figure, where probably the maximum loads are acting. For these stations the forces and moments are calculated for the whole variety of possibly critical manoeuvres (flight/landing conditions, A/C configuration and mass etc. as parameters) resulting in at least one loads envelope for each monitor station. Fig. 1.2.5-2 illustrates the concept of loads envelopes for the frontfuselage and the wing root. Indicated at the corner points of the envelope are the essential conditions, which lead to the load case.

As a first and in many cases correct approximation the design cases can be selected from the corner points of the different loads envelopes.

Usually there is a rather unique relation between corner points of a loads envelope and the flight parameters involved. Therefore considering modifications in the A/C's role or changes in equipment, mass or performance it is often straightforward to draw conclusions with respect to component load changes and therefore to stress/fatigue. This aspect is discussed in chapter 2 and 3.

1.2.6 Steps to Loads

In the foregoeing special aspects of determining design loads were discussed. To illustrate the practical sequence of steps to be carried out in order to calculate a flight load at a certain structure component the (simplified) procedure could be as follows:

1 Define mass and c.g.
2 Define point in Mach-Altitude range
3 Define sort of manoeuvre (symmetric, roll man., combined man. etc.)
4 Simulate manoeuvre and calculate response parameters
5 Calculate external net loads (forces & moments) on component from aerodyn. pressures, inertia forces etc.
6 Convert external load distribution to nodal point loads on FE grid
7 Analyse structure and determine local stresses (e.g. NASTRAN)

See also Fig. 1.2-1

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[1] net load = aerodynamic load + inertia load

1.3 Impact of Changes (Mass, Role, etc.) on Component Loads, Examples

Forces acting on an A/C caused by various effects:

Load	dependant on (list not complete)
Aerodynamic loads	incidence, sideslip, control angles, Mach, Altitude etc.
Inertia loads	Nx, Ny, Nz, angular rates and accelerations etc
Engine thrust	Mach, Alt, Combat thrust, idle etc.
Internal loads, e.g. cabin pressure	Specs, local accelerations
Actuator forces for control surfaces	Hinge moment =f(Mach,Alt)
Hydrostatic pressure	Local accelerations

The different kinds of forces and moments contribute to the loads on the monitor stations in a different manner. So the *front fuselage* up bending is clearly dominated by *inertia loads*, therefore an increase in the front fuselage mass will result in a higher front fuselage load, see Fig. 1.3-1

This is not an fictitious case, Tornado front fuselage mass has increased during the years and so the current critical load is definitely higher (max 15 %) than calculated during design.

In a similar manner it can be seen that the *rear fuselage* monitor station is dominated by inertia loads for the vertical bending, but aerodynamic loading (mainly from the horizontal tail) increases the total load, in contrast to the front fuselage case. Torque, which is unimportant for the front fuselage design, plays an important part for the rear fuselage and is almost entirely dominated by aerodynamic forces from the taileron (differential tail) and the fin (sideslip and rudder, horizontal gust), which may result in high loads during rapid roll manoeuvres.

Looking at the *wing*, it is clear that the wing bending is dominated by aerodynamic forces - the wing has mainly to carry the A/C weight - but substantial relief is effected by inertia forces as shown in Fig. 1.3-2. As indicated in the Fig., for the Tornado A/C the wing root bending moment is by 11% less carrying outboard stores than for the clean wing. If therefore the assumption for fatigue design, that the Tornado A/C is predominantly flown with stores on the outboard wing station, does not correspond to reality, a severe reduction in lifetime should be taken into consideration. This example highlights very clear, how changes in the usage of an A/C affect lifetime and how this can be assessed by rather simple considerations.

The following case of the Tornado *undercarriage* may also highlight the impact of how design loads are calculated and how usage assumed during design may be completely different from real life usage:
When it became apparent that the number of starts and landings for a certain Tornado squadron was much higher than projected the conclusion was that the lifetime of the squadron A/C was exhausted, at least with resp. to the landing gear. The question arose, whether lifetime could be prolonged and an investigation came to the following conclusions:

- Design of the landing gear was based on the assumption of dry runway conditions. But in reality dry runway landings occured much less than expected. Dry runway landing yields higher loads because of an high friction coefficient. Conclusion: lifetime can be extended
- At the same time takeoff and landing mass had increased, causing a lifetime reduction.
- Assumptions during design that approximately 50% of all landings would be 3-point landings were completely unrealistc - from which a premature damage resulted during Tornado MAFT. As only about10% of all landings were identified to be 3-point landings, the nose landing gear could be expected to have a far longer lifetime than projected.
- Overall methods (e.g. MIL) often result in safe but unrealistic loads. A rational analysis (simulation) of landing led to more accurate loads and therefore to a far better assessment of landing gear lifetime.

Considering all the points together sufficient rest life could be guaranteed.

1.4 Qualification of Loads, Static and Dynamic Tests

Static and dynamic loads critical for the structure are checked not only during the early stages of aircraft operational flight test but previously through ground tests as required by the certification procedures for the individual aircraft type.

The major milestones for ground testing are the ground resonance Test (GRT) to check dynamic structural response and confirm flutter margins established analytically to prevent flutter during initial flight tests, the "Major Airframe Static Test" (MAST) and the "Major Airframe Fatigue Test" (MAFT) for critical loadcases identified during structural analysis. The loads for both tests coincide with the loadset used during the development phase, a requirement critical for validation of analytical results.

One possibilty to prove the correctness of loads itself can be done by wind tunnel measurements (pressure plotting wind tunnel model or component balances) and/or modern flight load survey. Flight load survey provides information from exact inflight pressure measurements which, together with wind tunnel data, is fed back to the aerodynamic model of the A/C and leads to an update of the Loads Model, including other reference data (masses etc.). Then critical load cases are recalculated and thereby confirm/update design load calculations. A typical layout of pressure measurement locations for flight test is shown on Fig. 1.4-1.

A further procedure to gather flight loads data is by measuring net loads with calibrated strain gauges.

2. AIRCRAFT ANALYSIS USING STATIC LOADS AND FATIGUE LOADS SPECTRA

2.1 Static load conditions and fatigue sprectrum generation

Safety of flight for any aircraft rely on the recognition that the structure must withstand maximum static loads as well as repeated loads in addition to a certain amount of manufacturing defects and in-service damage throughout the service life without detrimental degradation of the structure leading to catastrophic failure of components. The two major tools for achieving this are the engineering analysis in accordance with the Structural Design Requirements (SDR) and fleet inpection programs.

The SDR documented in the aircraft weapon systems specification are the background for the set of loadcases to be adressed during the sizing of the different aircraft components. In general these loadsets can be devided into the following groups:

* Limit loadcases	(relevant for fatigue design requirements)
* Ultimate loadcases	(relevant for static strength requirements)
* Special loadcases	(i.e.birdstrike,crash,weapon release, buffett, etc.)

The defined set of missions for the aircraft configuration is the base for the generation of static and fatigue loadcases, which the structure should withstand throughout its intented service usage under defined environmental conditions, demonstrated through engineering analysis in the development phase and proofed via full scale testing (static ultimate and fatigue) later. Typical static loads criteria for a "care free handling"-flight control system equiped aircraft are shown in Fig. 2.1-1.

The results of the calculations are documented in "Static Strength Reports" for each part and form the input during the flight envelope expansion phase from the structural side, the socalled "Strength Envelope".

Durability or fatigue criteria are extracted from the planned/defined mission profile and combined with the overall life requirements in term of flight hours (FH) and/or flights within a defined timeframe of service years. If several A/C-roles are defined in the specification, overall life is split into Flights/Mission, appropriate representation of fatigue critical conditions within the fatigue spectrum is essential.

Maneuver loads are covered by an "overall g-spectrum" for the prime A/C-missions, i.e. Air-to-Air or Air-to-Ground as "Points in the Sky" for a given Mach/Altitude level and A/C-Weight/Store-configuration. Exedance curves are then generated as shown in Fig. 2.1-2.

Special load spectra are needed for components like control surfaces, airbrakes, engine mounts, stores or landing gear. For transport A/C cabin pressure cycles are an important factor for fuselage durability together with gust spectra.

The various loading spectra form the basis for the fatigue or fracture mechanics analysis depending on the design concept -*Safe Life* or *Damage Tolerance*- adopted.

2.2 Conversion of "external loads" into structural airframe loads

For the static and dynamic analysis of airframe structures a mathematical model of the aircraft is build using the Finite Element Analysis (FEA) -technique, representing the geometrie and structural stiffness of the major items and providing the bases for generation of "internal" structural forces in components like bulkheads,longerons, skins, spars and ribs etc. as well as other important information like maximum deformation of parts under loads.The detailing of these FE-models depend on the different phases within the iterative process and has improved dramatically with computer performance and modern Pre- and Post-processing capabilities in recent years. "Global" models are used to perform the global load path i.e. total aircraft or large component models. "Local" models in general are more detailed and they do simulate the special stiffness distribution like thickness changes, cut outs etc. Structural trade-off studies with this techniques in all phases of airframe development are standard procedures for some years, computer based optimisation of major elements like skin thicknesses are used today in early design stages. A decrease of computer cost and processing time, and in parallel the improvement of model generation, linking the design software (i.e. CATIA) with the loads model output of FEA-nodal forces and the finite element solver through preprocessors, will continue this trend towards more detailed models, better (and more) pre/post-processing information but also increased number of loadcases and refined component loads as discussed in chapter 1.2.

Fig. 2.2-1 shows a typical "coarse mesh"-finite element model of a wing structure with wing box and flaps, where 40-50 "design loadcases" were identified from the loads database of 500 load conditions and used for subsequent strength analysis. Fig. 2.2-2 shows a similar model of a center fuselage for a fighter aircraft.

The general trend in international programs towards development and production-workshare is mirrored in the global finite element model as well through superelement techniques requiring detailed data transfer checks and-protocol requirements. The EF2000 global model shown in Fig. 2.2-3 was generated by 5 european aircraft companies on different computer hardware and operating systems, therefore model compatability and -quality checks were essential during the socalled "Check Stress Full A/C- Finite Element Modell Static and Dynamic Assembly". The overall model size is about 35000 elements and more than 580 loadcases after superposition. After the unified analysis the results were transferred back to each company for further processing and structural analysis.

To further detail the loads in components and individual parts for actual sizing of the structural members, a "cut-out" of the global model with the exact boundary conditions applied to the "edges" of the component of interest from the results of the global model is possible and often used for detail investigations like effects of local cutouts, reinforcements, stability checks, etc.

Fig. 2.2-4A and 2.2-4B shows an example of this technique for a center fuselage bulkhead.

The results of these detailed model technique provide the background for strength analysis of static ultimate loads as well as fatigue loadcases in accordance with the allowables for the materials used and the geometric effects in the design.

3. LOADS MONITORING AND "FATIGUE LIFE" OF AIRFRAMES

3.1 Historical Overview

Fatigue management requirements and techniques have evolved over a period of more than 40 years, originating from simple cg-acceleration-counters to multi-channel systems with on-board processing capabilities. Originally a driving factor for load measurements was the generation of databases for design purposes, especially the wing loads and the wing to fuselage interface was of interest for subsonic and aerodynamical stable A/C-configurations. Combining the data with parameters,easy to retieve like speed, altitude, weight and time this transformed later into the bases for a first set of "fatigue meters", used as a tool to record repeated service loads on the airframe.

During 1960 and 1970 the fact that loads on many parts of the structure could not be related in any way to cg-acceleration and the simplyfied approach of the fatigue meters led to improved methods of fatigue monitoring. The first approaches to monitor on a fleetwide basis evolved and the philosophy of monitoring local fatigue sensitive areas, using mechanical strain recorders Fig. 3.1-1, or calibrated strain gages on the structure were introduced to record strain histories and calculate fatigue damage. In 1968 the NATO Military Committee required a SMP-Study on "Fatigue Load Monitoring of Tactical Aircraft" which subsequently presented agreed conclusions and recommendations for efforts to:

* Establish statistical relationships between movement parameters and structural loads
* Develop simple strain recording techniques
* Establish fatigue life monitoring techniques for all NATO countries

Within the last two decades a number of concepts for aircraft loads monitoring with either fleetwide data recording, supplemented by additional data from limited number of aircraft representative for squadron usage or individual aircraft tracking methods have been developed (1).

3.2 Loads Monitoring and Damage Rate Assessment

Monitoring of the airframe loading szenario and technologies to assess the "Used Life" or "Damage Rate" of airframe structures is a key element to the management of an aging aircraft fleet. The term Aging Aircraft can be defined in many different ways, among them are flight hours (or equivalent flight hours) approaching the designed service life; number of flights reaching the designed number of ground-air-ground cycles; or even pure age in the form of calender years. From a structures point of view the governing factor for aging airframes is the degradation of strength and ridigity of structural components with time and usage, applied to the aircraft as damage of different nature, the most obvious ones beeing fatigue cracks and corrosion. This degradation will continue, increase and finally form a threat to safety of flight without appropriate actions in the form of prevention, detection and repair through scheduled maintenance efforts.

Therefore the term "Damage Rate" has been identified as an indicator for the structural status of an aircraft, where a damage rate of 100% identifies the end of the fatigue life of a component or the limit for economic repair and usage of the aircraft.

3.2.1 The Object of Fatigue Monitoring Programs

In service individual aircrafts are subject to different operational loadings causing different damage rates in their fatigue prone areas. Dependent on how an aircraft is used, it may have an expended life significantly different from what is predicted at the time of service entry. The simple fact is that aircraft are often not used the way they were intended to be used during design and aircraft are used differently even when flown for similar missions. Fig. 3.2.1-1 shows an example for consumed fatigue life of a TORNADO lower wing skin with comparable missions and identical flight hours, Fig. 3.2.1-2 the wing root bending life-consumption for CF-18`s from one squadron. Factors of up to 5 for the damage rate have been identified between the most and least severe flown aircraft. If no fatigue monitoring program is carried out, maintenance actions, modifications and finally retirement of the equipment is based on the number of flight hours which the most severe flown aircraft is allowed to accumulate.

Hence, a sound and comprehensive operational loads data aquisition and evaluation will be an effective tool for cost savings during the operational life of an aircraft. With consideration of the life already consumed and with predictions about further usage the remaining service life of components can be determined and actions to adopt fatigue enhancement policies can be initiated at least for loads initiated damage, i.e. aircrafts with high damage rates can be allocated to fly less severe missions/configurations or structural modifications can be introduced before widespread fatigue damage occurs. Any monitoring and fatigue assessment program is therefore set up to answer the question: "What is the fatigue life ratio of the operational stress spectrum rated against the design/test spectrum on the different airframe locations?", or: "How many operational flight hours are equivalent to a simulated flight hour during fatigue testing ?"

3.2.2 Structural Monitoring Concepts and Systems
The main activities during a structural monitoring concept to determine the consumed life of each individual airframe are shown in Fig. 3.2.2-1.
The initial step of *Loads and Component Data Aquisition* is performed using flight data recorders for overall aircraft load parameters and local sensors for fatigue critical areas together with aircraft indentification information ("Tail-No.-Tracking") or component information for exchangeble items (i.e. horizontal stabilators).
Special post-processing is needed to separate, correct or replace faulty data.

The *Damage Calculation* is performed with repect to the design philosophy of the aircraft:

* For Safe Life - structures the calculation is based on S/N-curves and Miners rule to determine the accumulated damage.
* For Damage Tolerant designed structures initial flaws are assumed and crack growth analysis is performed for each fatigue critical part of the structure, ensuring that the initial flaw of a given size (i.e. 0.005 in or 0.127mm) will not grow to a functional impairment size within a given lifetime. Inspections, replacements or repair actions are scheduled by durability analysis using the flight loads data in the form of cycle by cycle stress histories coupled by propability of dectecton (POD) data.

From the registered loads data, a Derivation of Standard Load Sequences or Spectra (SLS) is extracted to create specific parameter or load hististories. They should fullfill the following criteria:

* The mean damage of the registered load sequence of individual A/C should be equa to the mean damage of the SLS
* The distribution of actual missions, configurations and other relevant operational parameters should be characteristic for the A/C operational usage. In some cases different SLS or spectra have to be generated for one A/C, i.e. Training-, Air-to-Air or Air-to-Ground dominated usage.

The *Fatigue Life Substantiation* is demonstrated through fatigue analysis and a qualification process including component and full scale fatigue tests in the development phase, validation of loads within flight envelope tests as well as operational experience during A/C-usage.
Since the tests are usually carried out within or in direct sequence with the design phase and based on the loads and structural configuration status of this time, deviations during the operational usage phase are normally scaled to the fatigue test SLS, determining the socalled "Usage Factor".

Assessment of the allowable fatigue life depends on the results of the fatigue life substantiation (in most cases the full scale test) and the design philosophy. Demonstrated fatigue test hours devided by the scatter factor and linked to the standard load-spectrum are the limit for the safe life designed structure, whereas for damage tolerant structures the test hours leading to cracks that impairs function of the structure devided by a factor are considered for the *Calculation of Fatigue Life.*

The *Consumed Life or Damage Rate* for each component is the relation of the actual individual A/C damage calculation and the allowable life and is used to schedule inspections, replacements or repair actions in order to ensure structural integrity.

3.2.3 Aircraft Tracking Systems for the GAF-TORNADO Weapon system
The TORNADO Multi Role Combat Aircraft was designed in the early ´70 and followed the safe life design principal for durability with a scatter factor of 4, used on the design life of 4000 FH. The fatigue tracking concept of the A/C is divided into three sectors with different numbers of A/C´s from the fleet involved and different amount of data (parametric and strain gages) gathered, as shown in Fig. 3.2.3-1.
Monitoring is based essentially on flight parameters, which are available through the existing flight recorder unit and defined as Recorder Parameter Set (RPS). An extendet Full Parameter Set (FPS) is generated through differentiations and conversions of existing data. The flight recorders are distributed on a statistically representative basis throughout the squadrons and register the spectrum of selected aircraft. Additionally, strain gages in various fatigue critical areas of the structure are monitored on a limited number of aircraft, the results are evaluated by regression techniques to produce a realistic correlation between operational strain on the structure and the flight parameters causing it.
A reduced Pilot Parameter Set (PPS) is collected from each individual A/C through the nz-counter plus aircraft weight and configuration data, Fig. 3.2.3-2 on a flight by flight bases.
Thus, a "multi-level" tracking is performed:

* Individual Aircraft Tracking with Pilot Parameter Set
* Temporary Aircraft Tracking with Recorder Parameter Set + Strain gages
* Selected Aircraft Tracking with Full Parameter Set

Fig. 3.2.3-3 lists the recorder parameter set and strain gage sampling rates for the Temporary Aircraft Tracking level.
From a conceptional point of view, the individual aircraft tracking permits optimum utilization of the structural life of a fleet. This naturally requires appropriate sensors existing in the individual aircraft for the aquisition of local stress history data. Since not all of the TORNADO-A/C are equipped with strain gages, PPS aquired by IAT

are converted via the regression table from TAT-A/C into stress spectra for the fatigue critical areas.
Monitoring of the TORNADO's fatigue critical areas uses the local strain concept, too. For this, a suitable local strain measurement location was established for every area during the Full Scale Fatigue Tests. Fig. 3.2.3-4 shows an example for a critical area in the engine duct skin, where a "reference" strain gage is located at the wingbox shearlink to the fuselage for on-board monitoring. The damage of this location is traced to the wing bending moment, applying the transfer functions for inner wing shear force and bending moment to the recorder parameter set and the correlation equation for the reference gage from fatigue test, the stress history for this area is generated.

3.2.4 On-Board Loads Monitoring System of Candian Forces CF-18 Aircraft (2)

Usage characterization of the CF-18 fleet is also a key element of fatigue life management of the CAF F-18 fleet. In contrary to the TORNADO, all of the CF-18 aircraft are equipped with strain gage sensors at different locations during production , Fig. 3.2.4.-1.
Flight Parameters are recorded together with the strain gage signals on a flight by flight bases within the Maintenance Signal Data Recorder (MSDRS) and allow individual aircraft tracking throughout the service life of every aircraft.
Location of the strain gages were selected by the manufacturer based on criticality of the structure, its accessibility and the degree of protection from accidental damage. Prime and spare gages are applied for redundancy. Use of the direct strain measurements inherently acounts for parameters like airspeed, altitude, weight, store configuration and cg-variations during flight. However, the acuracy of the fatigue calculation is dependent upon the reliability and proper installation of the sensor.
Data are stored on magnetic tape and downloaded to a ground station. Different level of data reduction and reporting can be generated from limited fatigue analysis codes at operating bases to assess severity of individual flights or mission profiles to anual reports for longterm trend analysis.
Since the F-18 was also designed to the safe life philosophy, fatigue consumption is calculated in terms of Fatigue Life Expended (FLE) against the 6000 FH life of the design usage spectrum. This linear relationship was established using the information collected during full scale fatigue test conducted by the manufacturer and is scaled for CF in-service usage and structural configuration changes between test article and fleet.
For the purpose of fatigue calculations, crack initiation was defined as formation of a crack of 0.25 mm or 0.01inches. The cracks usally originate at locations of tensile stress concentrations, where material yield strength is exceeded when high load magnitudes are encountered in-service.
From the inflight MSDRS recorded strain peaks and valleys, a representative loading spectrum is generated, and by using the individual material stress-strain relationship of the components, the corresponding stress spectrum is obtained.
From this spectrum the amount of damage per cycle and afterwards the crack initiation life can be calculated by using material dependent S/N-curves, Fig. 3.2.4-2.
The FLE is then expressed as the total damage accumulation to date divided by the total structural fatigue damage required to initiate a 0.25 mm crack under the design loading spectrum.
After initiation, remaining life of the component is used by crack growth up to the critical crack length. Currently, the fatigue analysis program does not contain a crack growth prediction model.
Together with fatigue awareness and control programs, reducing configuration severety for missions, within 2 years of implementation, the CF was able to improve fleet attrition trends already by approx. one year of service, Fig. 3.2.4-3
Some of the experiences with the system of individual aircraft tracking through strain gage sensors are:

* Fatigue damage calculations are improved by direct strain measurements due to elimination of A/C flight parameters from the equations
* Accuracy of the measurements are vital and gage drift over time is a concern
* In flight-calibration of gages through reference maneuvers during maintenance test flights can be a solution to gage drift
* Reliability of the strain sensor is vital, since drop-outs must be filled with conservative "fill-in"-algorithmen, leading to artificially higher FLE data.
* Timely reporting schedules are essential for feedback of damage accumulation and on the effects of role changes/aircraft usage to the operational squadron as well as to the fleet manager.

4. INFLUENCE OF THE STRUCTURAL CONFIGURATION STATUS

An aircraft in service or produced over an extended period of time will change its structural and system configuration in many areas due to structural modifications, additional systems installed, improved engine performances etc.
While major structural modifications are usually covered by either extensive analysis, accompanied by component testing and sometimes even full scale tests, the smaller modifications and "updated" system installations are well documented in production configuration control files, but mostly "neglected" for internal loads influence for some time.
Fig. 4.0-1 shows the increase of the TORNADO structural mass aft of the rear transport joint at X12737, including vertical and horizontal tail for the different batches within a production period of 14 years together with the design weight used in the unified analysis in 1976.
The "immediate solve" for weight increase of reducing

internal fuel and keeping the nz-level (nz x m = constant) will obviously not work for this problem, based on the fuel sequence the wet wing mass definition is no longer valid and leads to higher wing loads. The same effect is also valid for the front fuselage, as explained in chapter 1.3.
At the same time engine thrust has been raised also by 16%, although only a fraction of it is used during peacetime operations, the heavier engine contributes to the mass increase. More important, in contradiction to a special role equipment, which may be cleared with restrictions like "Not for peacetime training missions", this mass increase influences the fatigue life consumtion permanently.
The influence of the higher loads can be clearly seen on the structural transport joint loadings leading to vertical shear increase of approx. 20 kN or 4500 klbs and vertical bending of 30 KNm or 265000 inlbs respectively additional 6.5 % on the design limit loads, Fig. 4.0-2.

A regular check of the present inertia loads status after modifications and system upgrades is therefore mandatory to make loads monitoring concepts, based on parametric data, work.

5. REFERENCES

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(2) M.B. Zgela and W.B. Madley, Durability and Damage Tolerance Testing and Fatigue Life Management: A CF18 Experience, AGARD CP-506

(3) H.-J. Baderschneider and K. Füllhas, "Update/Structural Status of TORNADO IDS&ECR Center Fuselage incl. Fwd Air Intake, DASA LME2-Study,1993, (Unpublished)

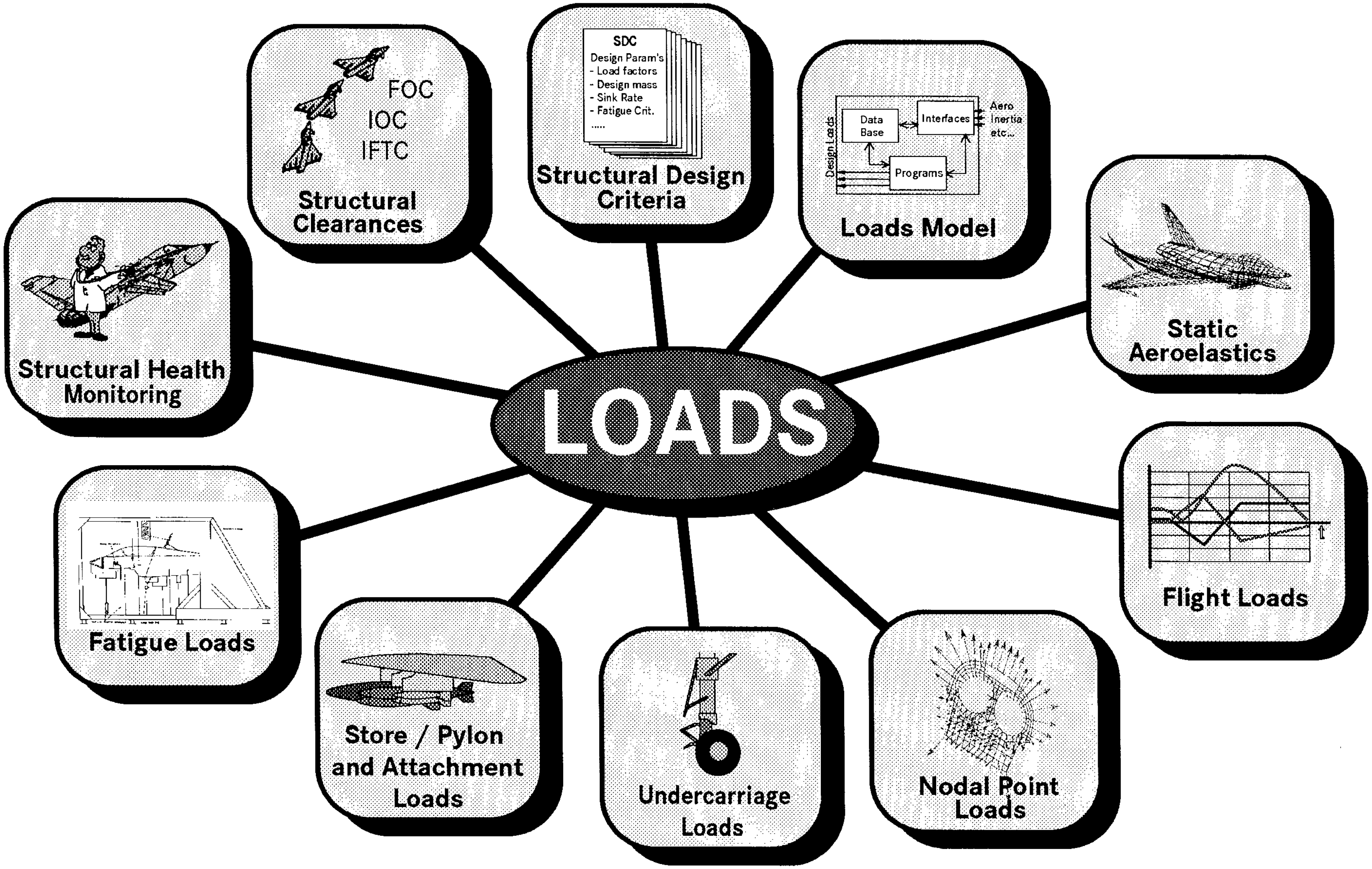


Fig. 1-1 Loads Main Tasks

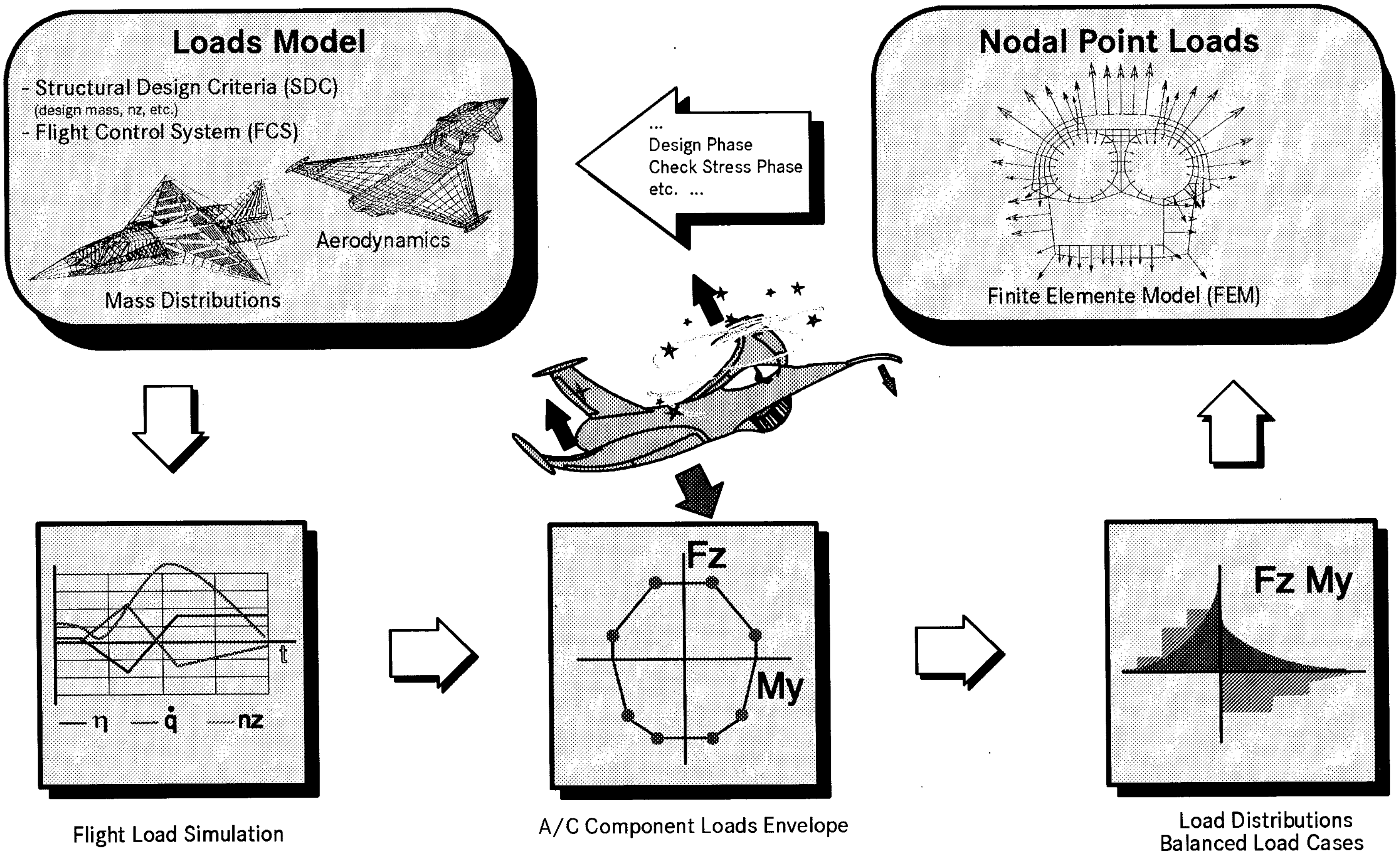


Fig. 1.2-1 Loads Loop

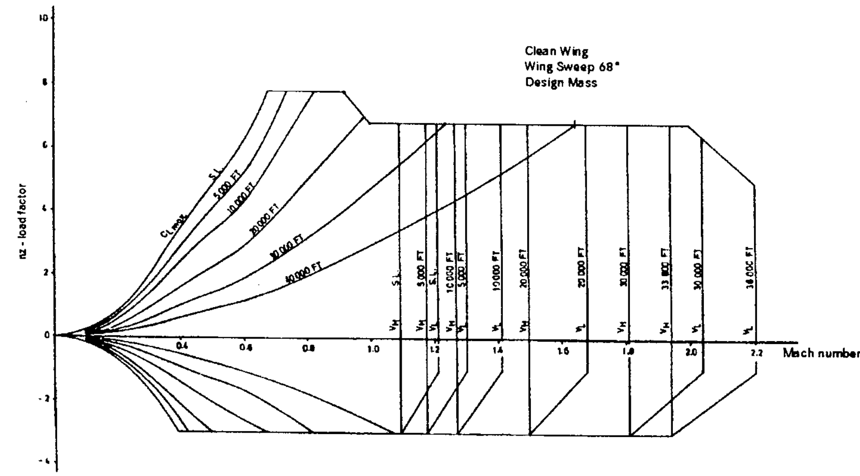


Fig. 1.2.1-1 Ma - n Diagram in Altitude

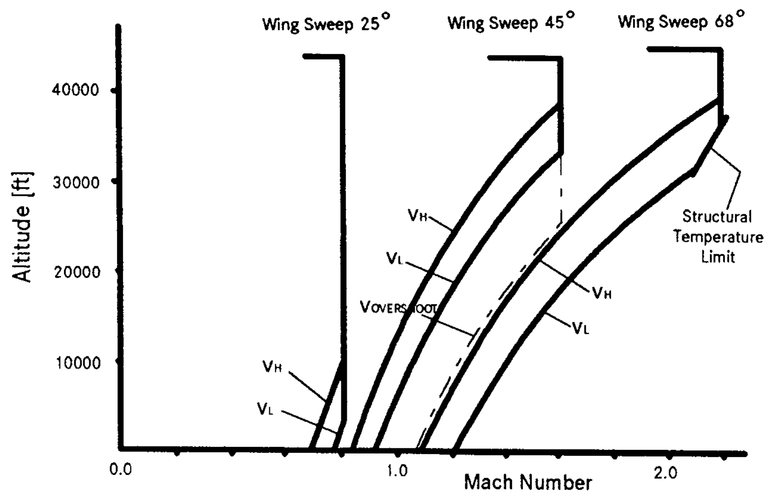


Fig. 1.2.1-2 Altitude - Mach Number Envelopes

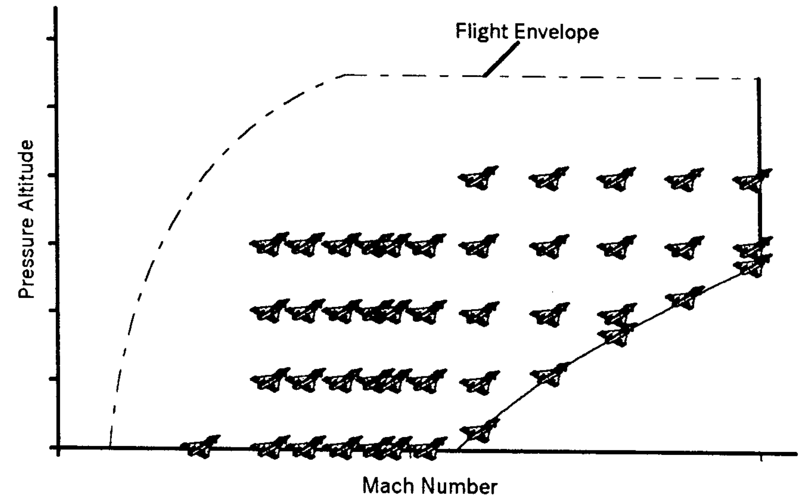


Fig. 1.2.1-3 Mach - Altitude Points of Loads Model (flex. Aerodynamics)

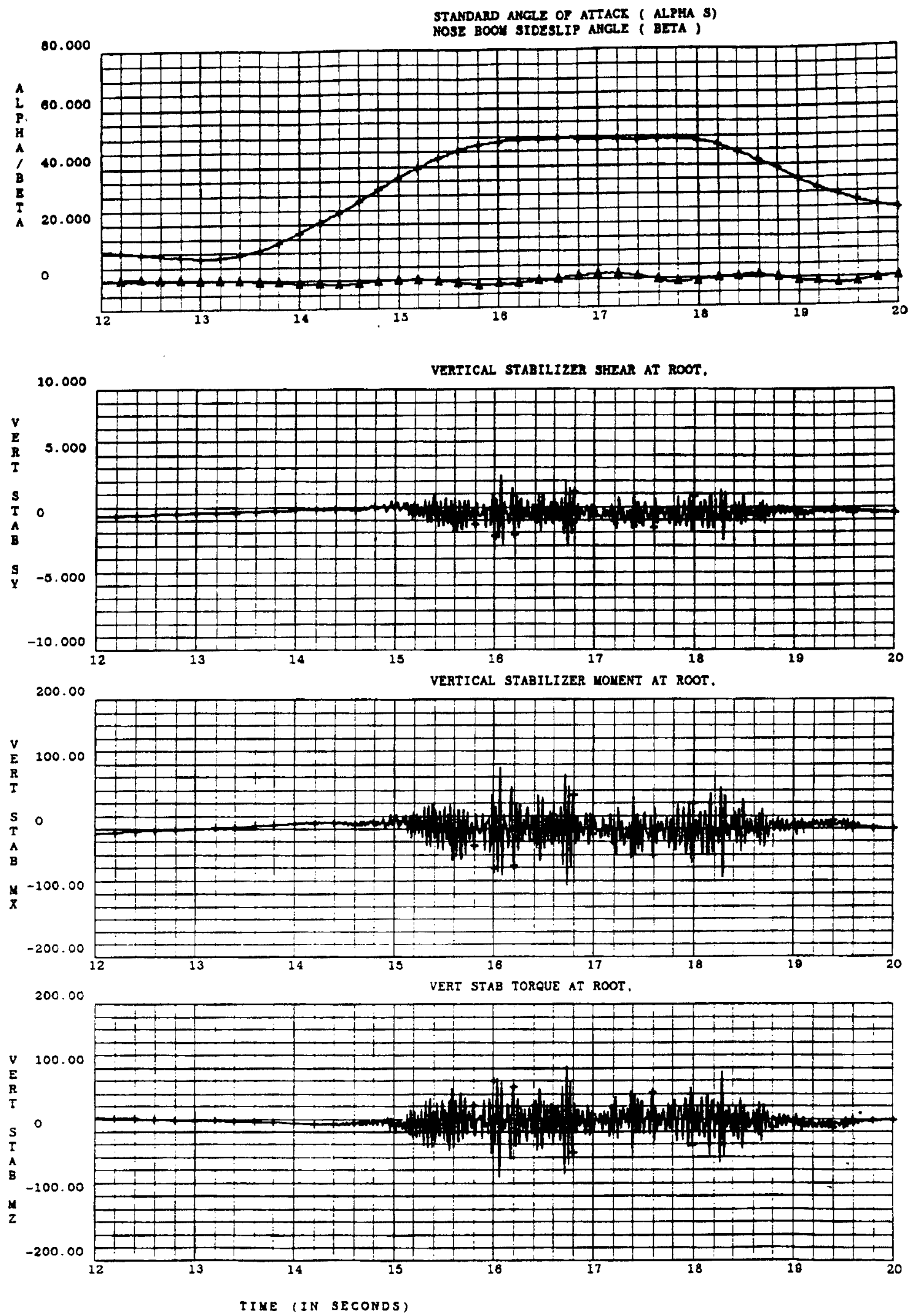
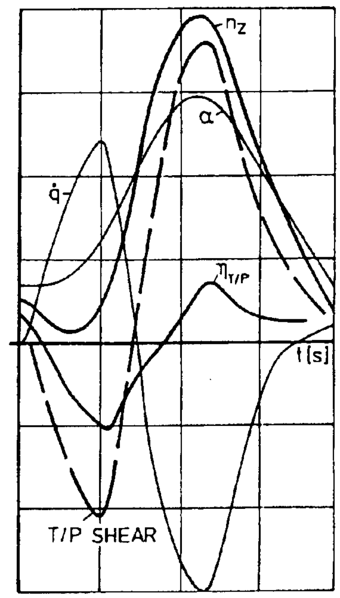


Fig. 1.2.2-1 Fin Buffet at High Angle-of-Attack (Flight Test Results)



nz	vertical load factor
α	angle-of-attack
$\dot{q}$	pitch acceleration
$\eta_{T/P}$	tailplane deflection angle
T/P SHEAR	tailplane vertical shear force

Fig. 1.2.3-1 MIL-SPEC Pitch Manoeuvre

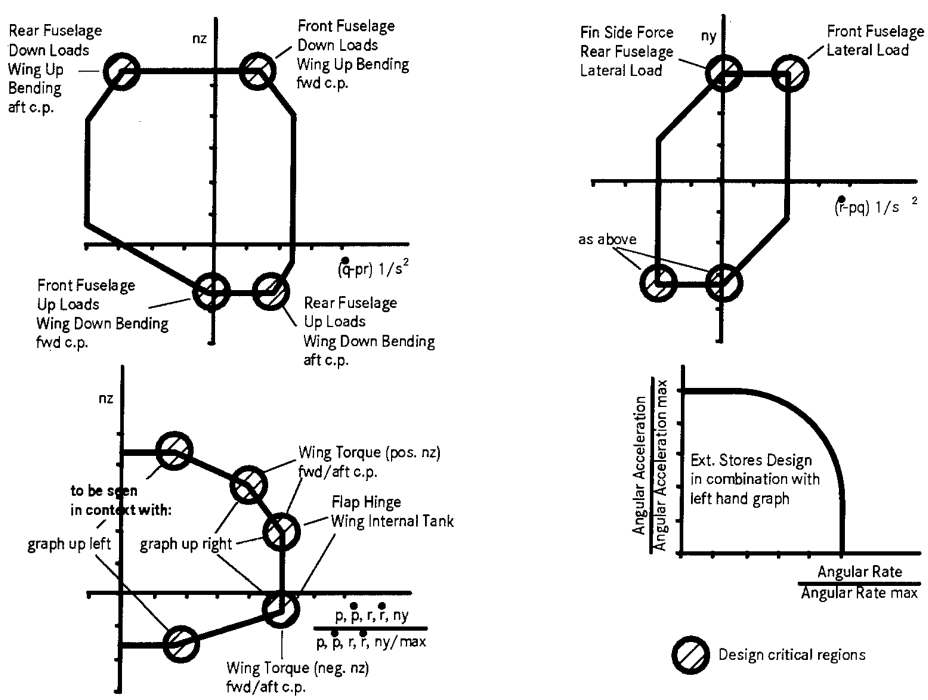


Fig. 1.2.3-2 Flight Parameter Envelopes for Structural Design

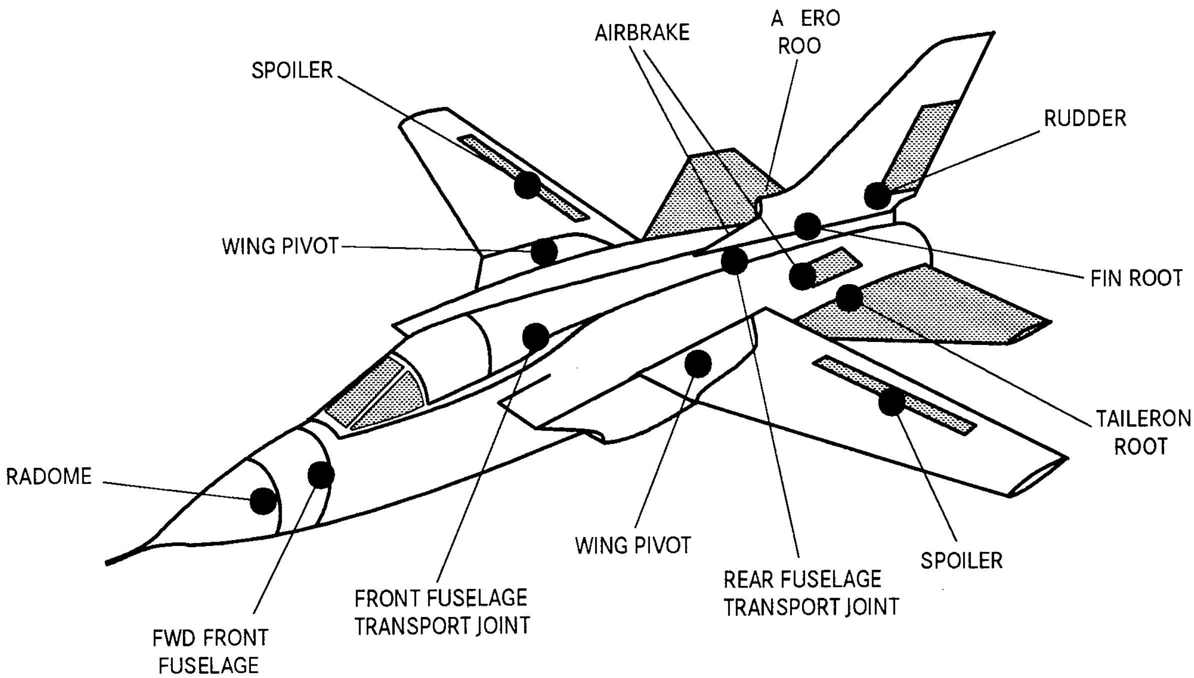


Fig. 1.2.5-1 Load Monitoring Stations

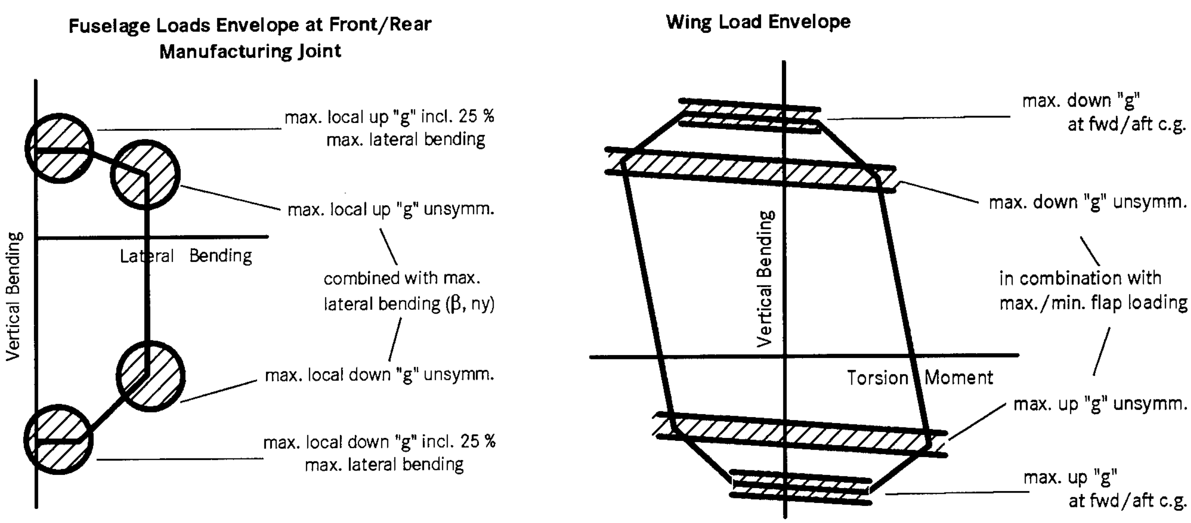
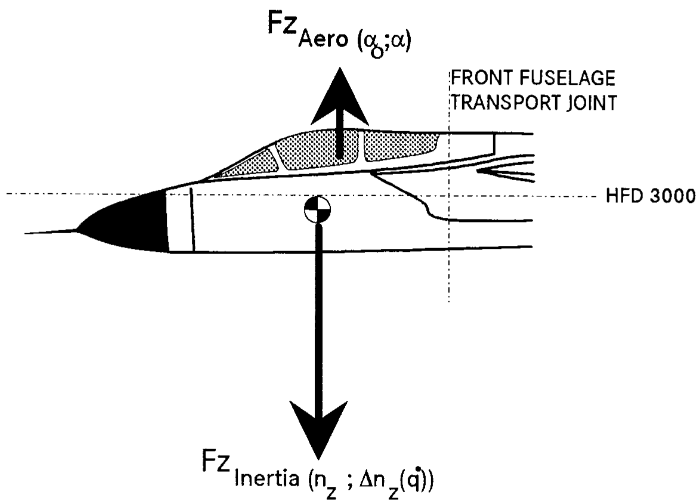


Fig. 1.2.5-2 Major Aircraft Component Loads Envelopes



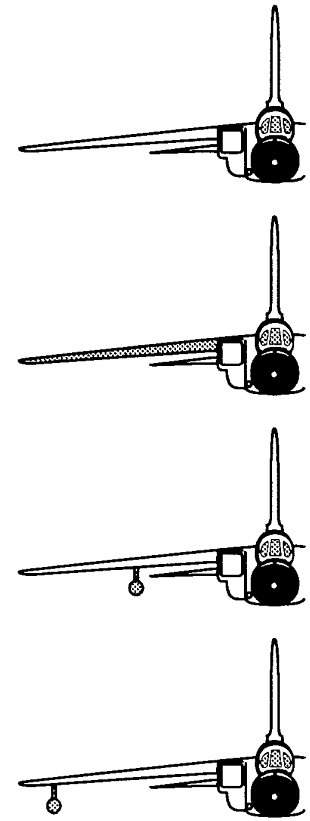
For the Front Fuselage the vertical shear force Fz and the vertical bending moment My are dominated by inertia load conditions.

Max. down bending design loads

	Air	Inertia	Net	
Vertical Shear Force	100	-310	-210	%
Vertical Bending Moment	100	-480	-380	%

<u>Conclusion:</u> An increasing Front Fuselage mass will lead to higher Front Fuselage loading.

Fig. 1.3-1 Front Fuselage Transport Joint Critical Load Conditions



Clean Wing - dry (Design Condition)

	Air	Inertia	Net	
Normal Force	*100*	*-15*	*85*	%
Torsion	*100*	*-13*	*87*	%
Bending	***100***	***-13***	***87***	%

Clean Wing - wet

	Air	Inertia	Net	
Normal Force	*100*	*-24*	*76*	%
Torsion	*100*	*-19*	*81*	%
Bending	***100***	***-19***	***81***	%

Clean Wing - dry + i/b store (350 kg)

	Air	Inertia	Net	
Normal Force	*100*	*-22*	*78*	%
Torsion	*100*	*-17*	*83*	%
Bending	***100***	***-17***	***83***	%

Clean Wing - dry + o/b store (350 kg)

	Air	Inertia	Net	
Normal Force	*100*	*-22*	*78*	%
Torsion	*100*	*-23*	*77*	%
Bending	***100***	***-24***	***76***	%

<u>Conclusion:</u> Adding mass to the wing (e.g. carriage of stores) leads to reduced wing loads.

Fig. 1.3-2 Influence of Wing Loading Conditions on Wing Loads

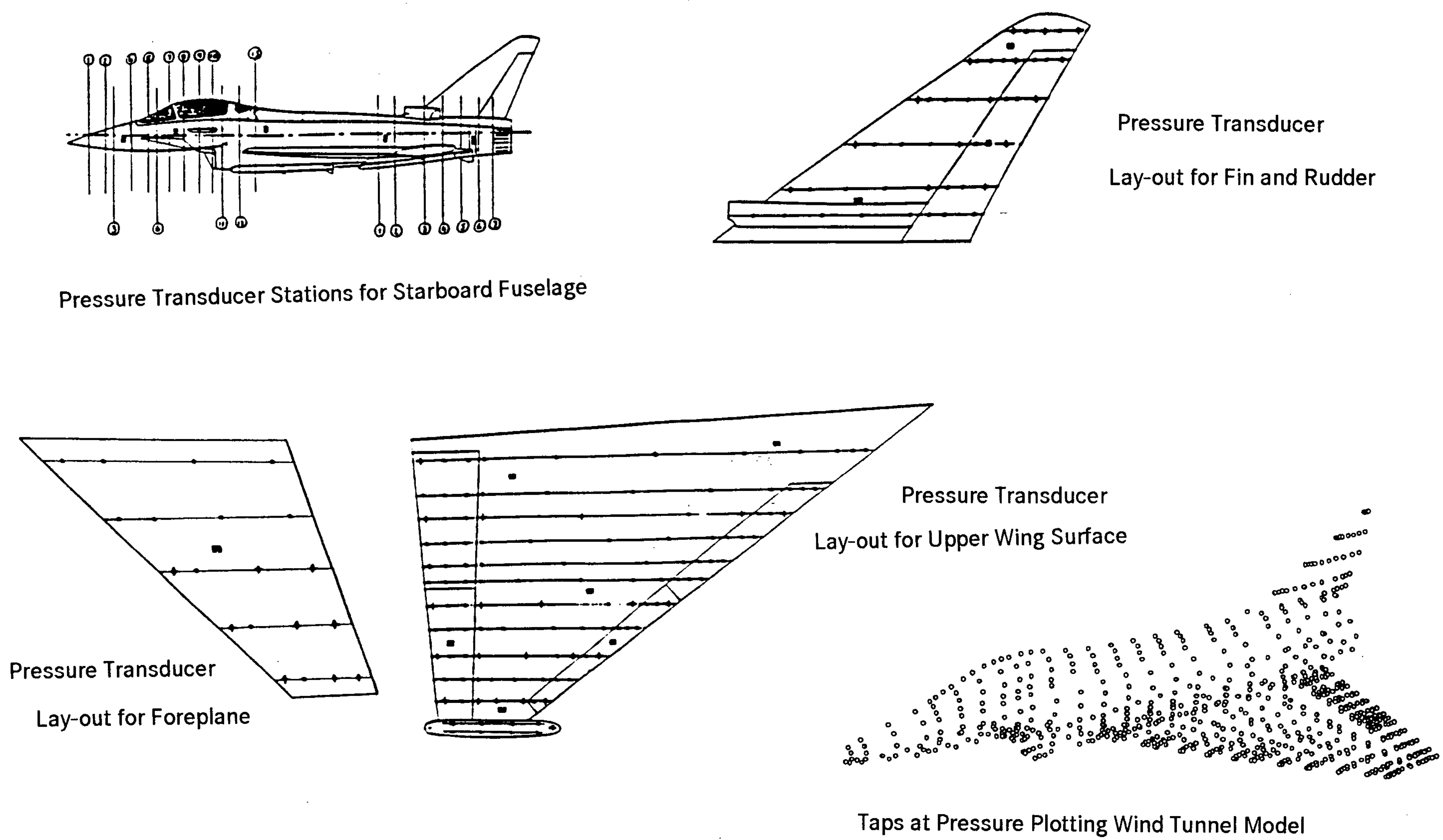


Fig. 1.4-1 Prototype Pressure Plotting for Flight Load Survey

STATIC LOADS DESIGN CRITERIA

- Two Load Levels: Design Limit Load (DLL) = Max. Operational Load in Service
  Design Ultimate Load (DLL) = Failure Load of Structural Components

- Ultimate Load: 1.4 x Limit Load
  for all Loadcases controlled by FCS

- Ultimate Load: 1.5 x Limit Load
  for all loadcases not controlled by FCS
  e.g. undercarriage cases, actuator loads, store attachments etc.

- Requirements: No structural failure at DUL
  No permanent deformation at DLL
  Buckling of panels must remain elastic at DLL
  No buckling at DUL for items where structural integrity is affected by stability
  No buckling up to 110% DLL for items where operational function is affected by stability

Fig. 2.1-1 Static Loads Design Criteria for Airframes

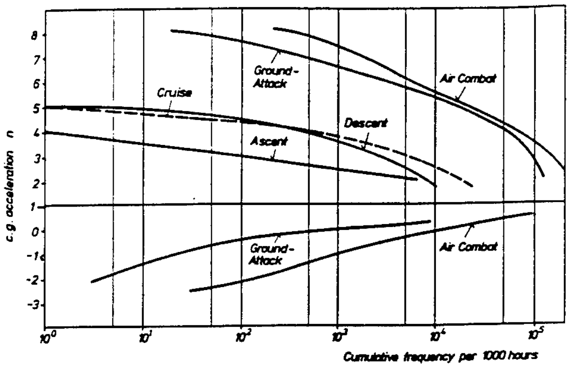


Fig. 2.1-2 Typical Excessdance Curves for Combat Aircraft

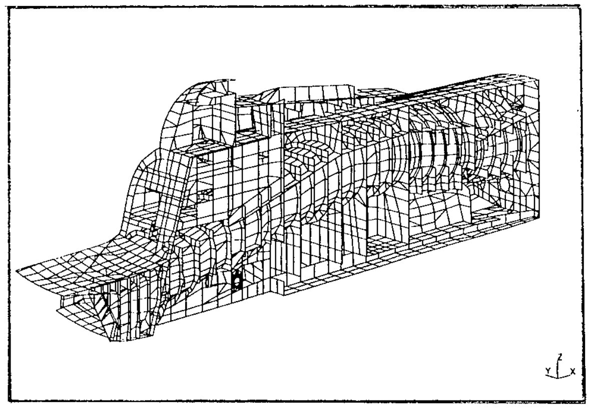


Fig. 2.2-2 FE-Half-Model of Center Fuselage Structure

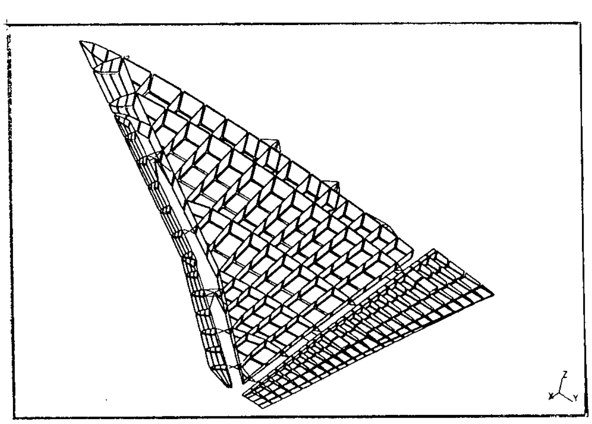


Fig. 2.2-1 Coarse Mesh FE-Model of Wing Structure

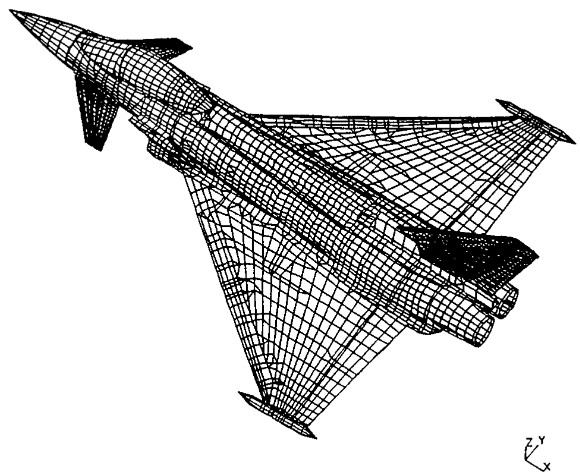


Fig. 2.2-3 EF2000 Global Model for Unified Analysis

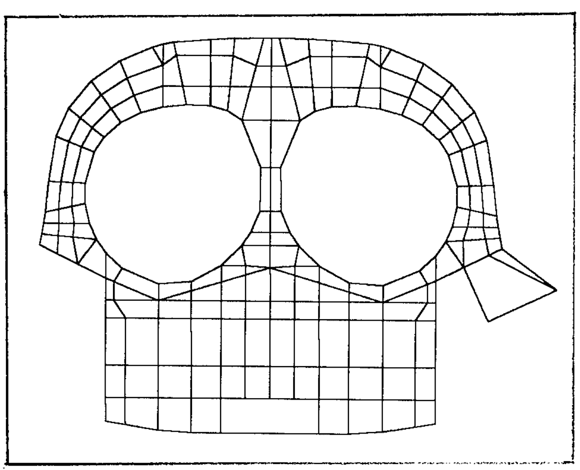


Fig. 2.2-4A Coarse Mesh FE-Model of Center Fuselage Frame

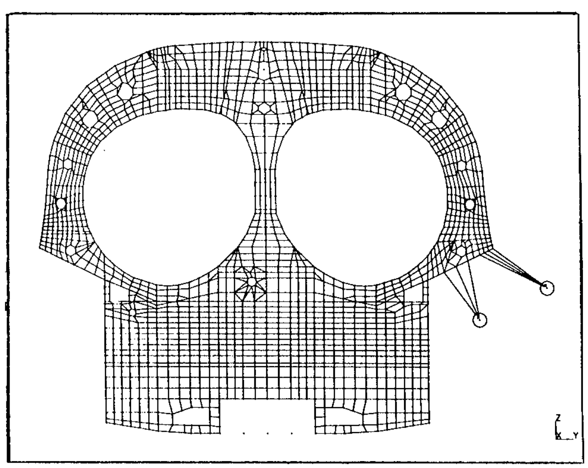


Fig. 2.2-4B Fine Mesh FE-Model for Detail Analysis

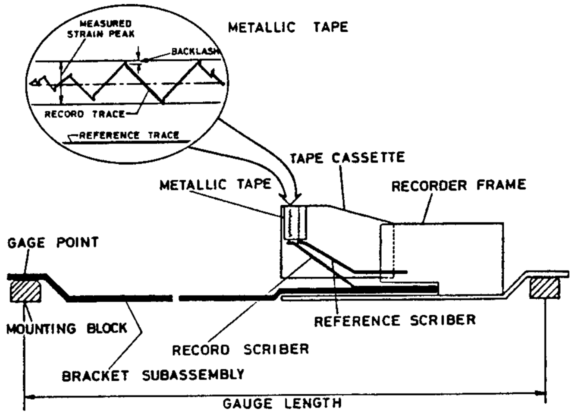


Fig. 3.1-1 Princip of Mechanical Strain Recorder

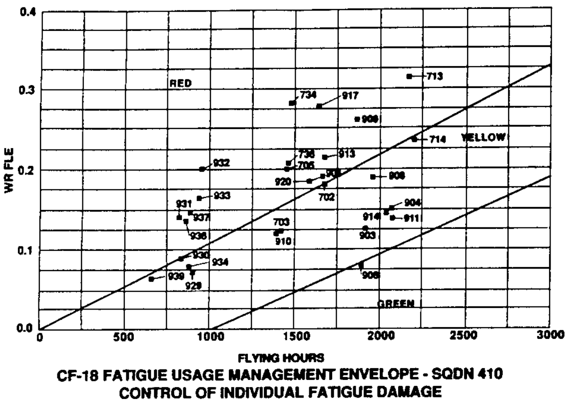


Fig. 3.2.1-2 Wing Root Bending Life Consumption, CF-18

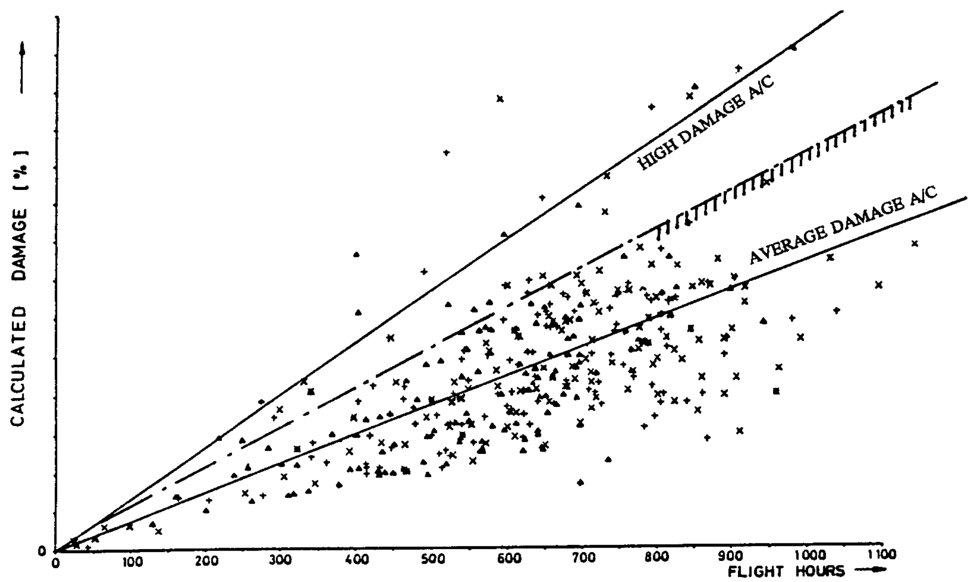


Fig. 3.2.1-1 Lower Wing Skin Life Consumption for Similar Missions, TOR

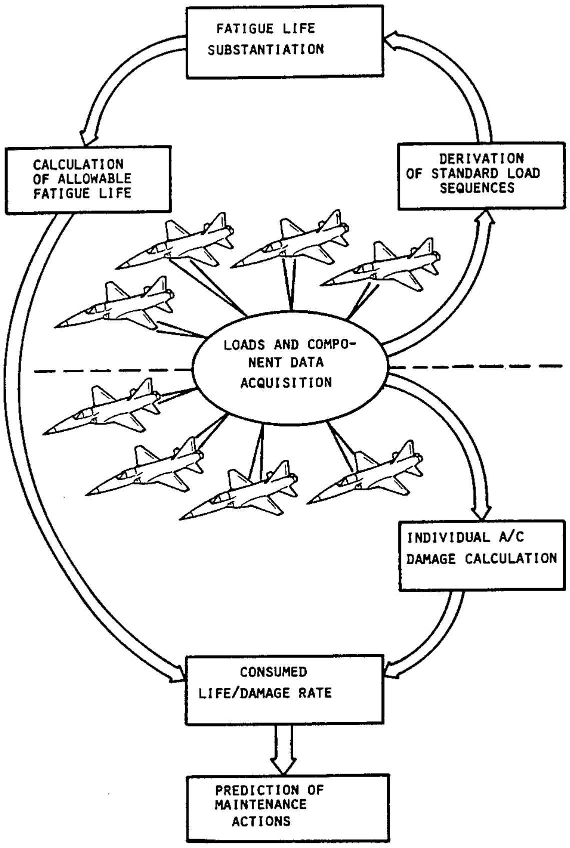


Fig. 3.2.2-1 Structural Monitoring Activities

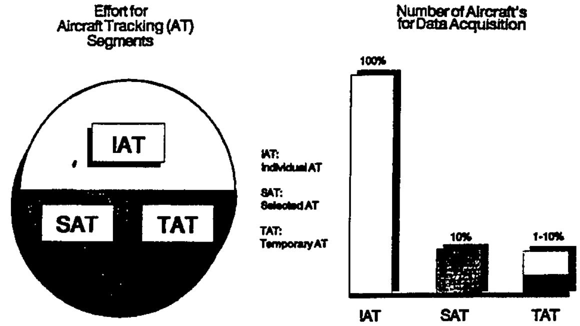


Fig. 3.2.3-1 Aircraft Tracking Segments, TOR

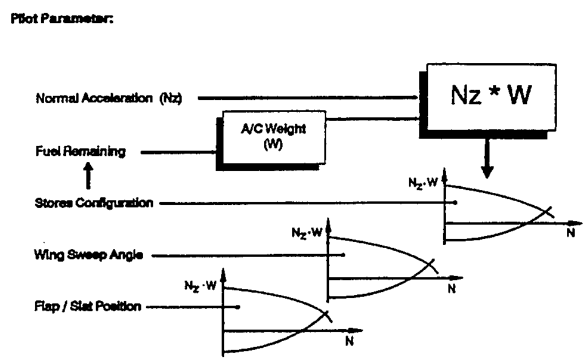


Fig. 3.2.3-2 Reduced Parameter Set (PPS) for IAT

No.	Parameter	Sampling Rate / s	No.	Parameter	Sampling Rate / s
1	Pressure Altitude	0.5	11	Inboard Spoiler STBD	1.0
2	Calibrated Airspeed	0.5	12	Rudder Position	2.0
3	Normal Acceleration	16.0	13	Wing Sweep Angle	0.5
4	True Angle Of Attack	2.0	14	Primary Strain Gauge	16.0
5	Roll Rate	8.0	15	Secondary Strain Gauge	4.0
6	Pitch Rate	4.0	16	Flap Position	1.0
7	Yaw Rate	2.0	17	Slat Position	1.0
8	Taileron Pos. PT	4.0	18	Fuel Remaining	1.0
9	Taileron Pos. STBD	4.0	19	Stores Configuration	4.0
10	Outboard Spoiler PT	1.0	20	Oleo Switch	0.5
			21	Identification Data	1.0

Fig. 3.2.3-3 Recorder Parameter Set Data and Sampling Rates

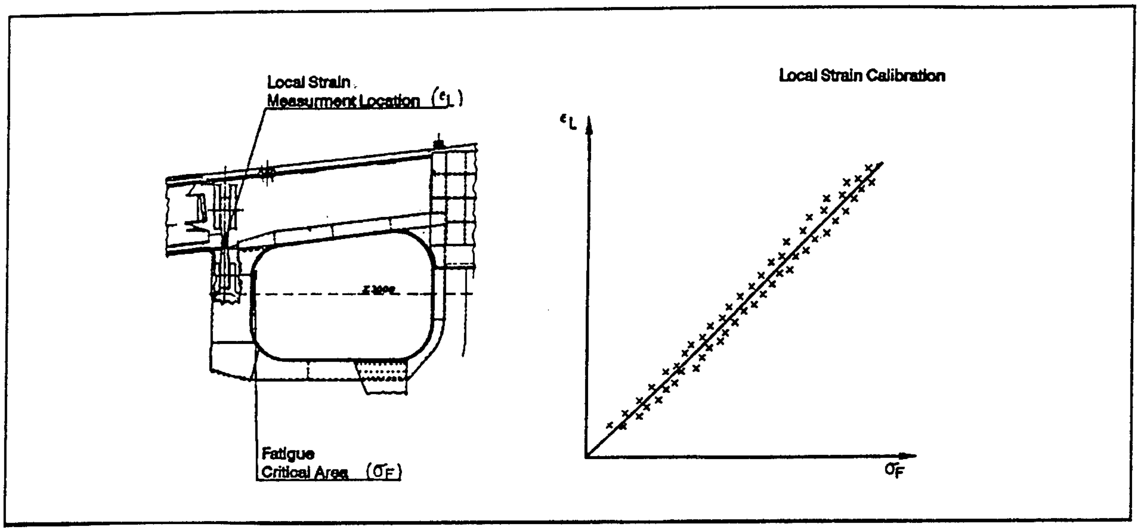


Fig. 3.2.3-4 Reference Strain Gage on Wing Attachment

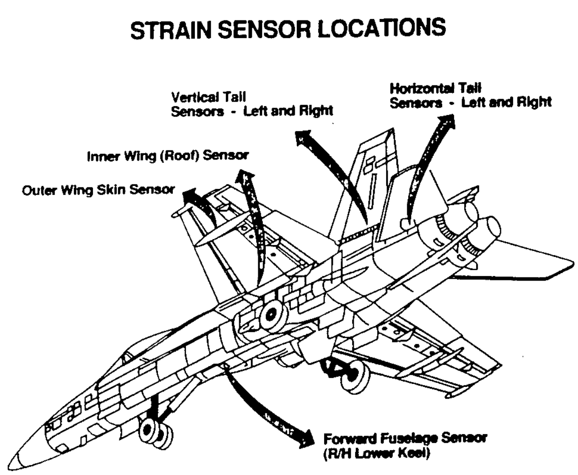


Fig. 3.2.4-1 CF-18 Strain Gage Locations

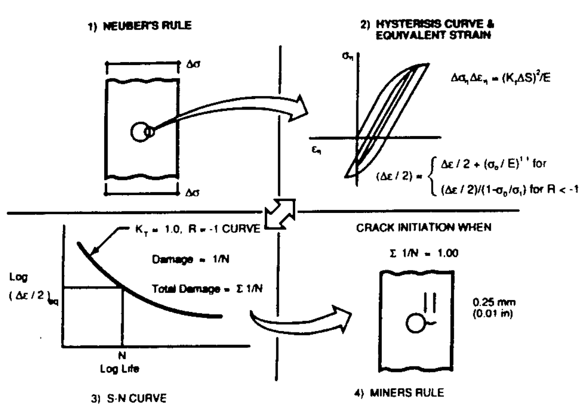


Fig. 3.2.4-2 Crack Initiation Concept

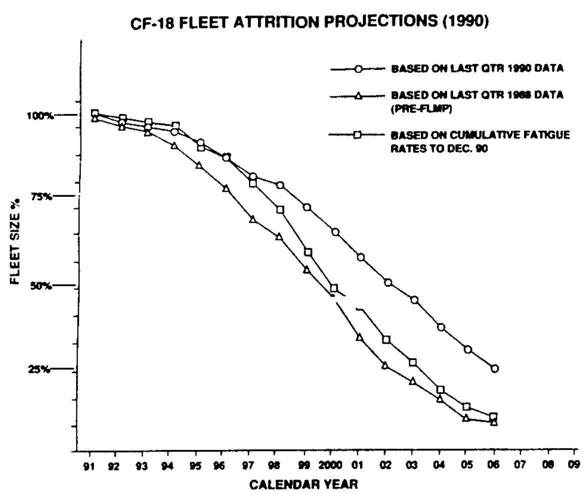


Fig. 3.2.4-3 Life Improvement of CF-18 Fleet

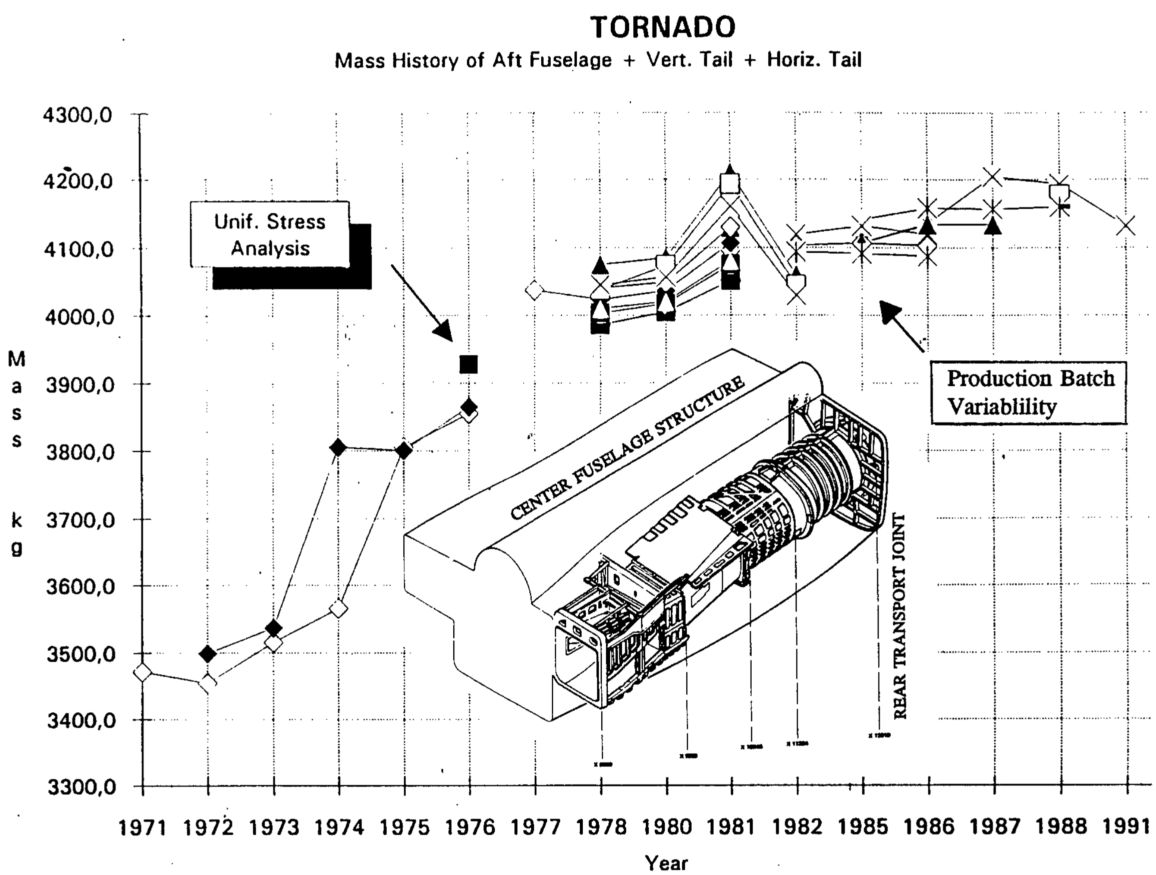


Fig. 4.0-1 Historic Structural Mass Increase of TOR Aft Fuselage

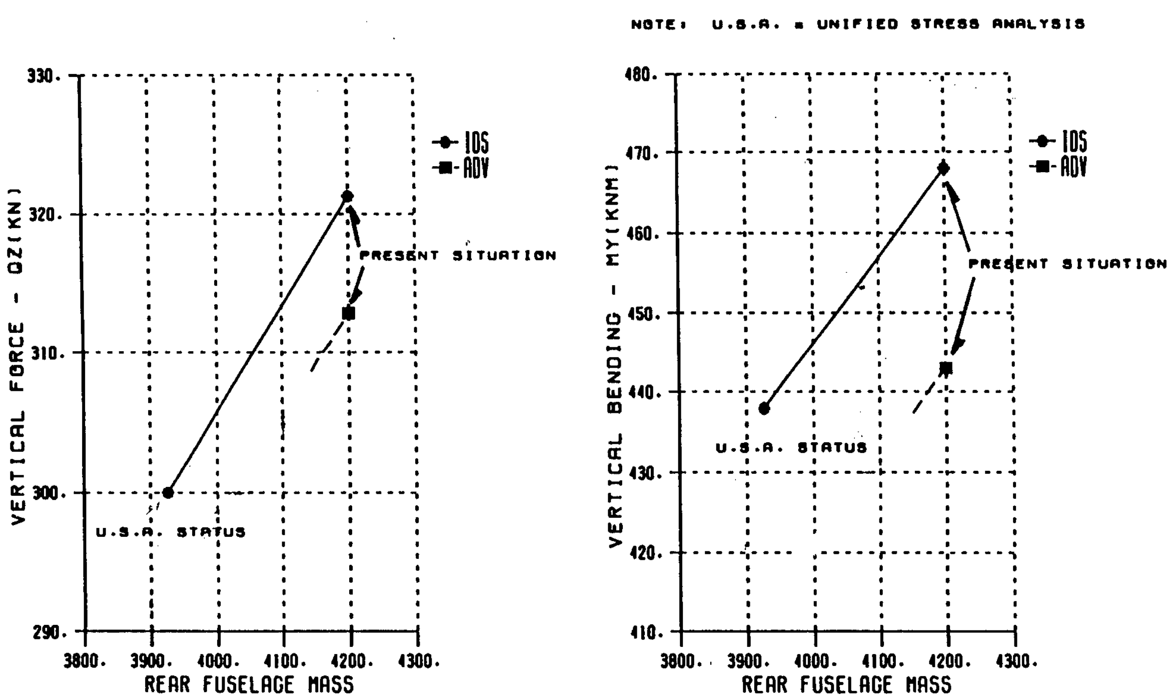


Fig. 4.0-2 Interface Load Increase at Rear Transport Joint

Corrosion Prevention System for the F-16 Fighter Aircraft

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SUMMARY

The Corrosion Prevention System for the F-16 Fighter Aircraft is an excellent example of the importance of defining the design, materials, and process selection requirements early in the acquisition program. This process necessitates working closely with the contractors and customers throughout the life of the program and establishing and maintaining a Corrosion Prevention Advisory Board (CPAB).

The corrosion prevention system used on the F-16 implements materials, surface treatments, finishes, and coating systems, that provide superior corrosion protection when manufactured and maintained properly. The F-16 has nevertheless suffered from some corrosion related problems. Although many of these problems have been corrected, field corrosion surveys are useful in identifying new issues as well as past problems that have not been adequately addressed. Results of a recent field survey conducted by the F-16 System Program Office (SPO), miscellaneous items from the CPAB meetings, and continuing changes brought about by environmental, health, and safety compliance provide the need for a strong, dynamic, on-going corrosion prevention program.

LIST OF SYMBOLS

HVLP - High Velocity Low Pressure
LANTIRN - Low Altitude Navigation and Targeting Infra-Red at Night
NDI - Non-Destructive Inspection
TO - Technical Order
TCTO - Time Compliance Technical Order

1. INTRODUCTION

The worldwide F-16 fleet comprises nearly 4000 aircraft, with distinct differences designated by "block" and model of aircraft (i.e., Blocks 1/5/10/15/20 are A and B aircraft; Blocks 25/30/32/40/42/50/52 are C and D aircraft). A and C models are single-seat aircraft, while B and D models are two-seat aircraft.

The F-16 fighter aircraft has seen its share of corrosion related problems, but compared with other United States Air Force (USAF) aircraft systems, the F-

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16's problems are minuscule. A large part of the F-16's success is due not only to the advances made in materials and coating technologies over the years, but also to the "up-front" planning and aggressive attention to corrosion prevention throughout the life of the F-16.

As figure 1 shows, the F-16 corrosion prevention system began with the Air Force Regulation 400-44, *Air Force Corrosion Program*, which "outlines the policy and procedures for managing an effective corrosion prevention and control program for all Air Force systems." The Aeronautical Systems Center (ASC) supplement to AFR 400-44 defines the responsibilities of Engineering (EN), Wright Laboratory (WL), and the System Program Office (SPO). The supplement stresses the importance of designing with corrosion prevention in mind and calls for a Corrosion Prevention Advisory Board or CPAB to be implemented "as early as practical during the demonstration and validation phase."

In order to make corrosion prevention a major consideration in design, the top-level specifications must define the requirements. The Weapon System Specification (WSS) defines the environmental conditions and identifies the Corrosion Prevention and Control Plan and Prime Item Development Specification (PIDS). The PIDS references the environmental criteria and the contractor's finish specification. As a result, all of these referenced documents are Type I specifications and require F-16 SPO Multi-national Configuration Control Board (MCCB) approval.

The Corrosion Prevention and Control Plan defines the contractor's Corrosion Control Team (materials, quality assurance, process control, etc.) and Defense Plant Representative Office (DPRO) duties, specifies the corrosion prevention and control system, and summarizes the contractor's finish specification.

The environmental criteria report defines the general climatic and environmental design requirements for the F-16 aircraft (i.e., temperature, pressure, humidity, salt fog, etc.).

The contractor's finish specification covers the detailed procedure the contractor follows in applying corrosion protection to the aircraft surfaces and component parts, including spares. This includes the methods and materials required for cleaning, surface treatment, and application of finishes and protective coatings.

The F-16 CPAB is funded via the F-16 Program Management Directive (PMD). The PMD directs development, production, deployment, maintenance, system support, and modification for the applicable aircraft. The PMD also directs the CPAB "to include representation from Air Force Materiel Command (AFMC), Air Combat Command (ACC), and Wright Laboratory (WL)." This statement in the PMD is the basis for funding any F-16 corrosion-related effort.

The CPAB charter defines the purpose, membership, responsibilities, and procedures for the CPAB. The F-16 CPAB has diverged recently into two separate meetings: An "A" meeting, still

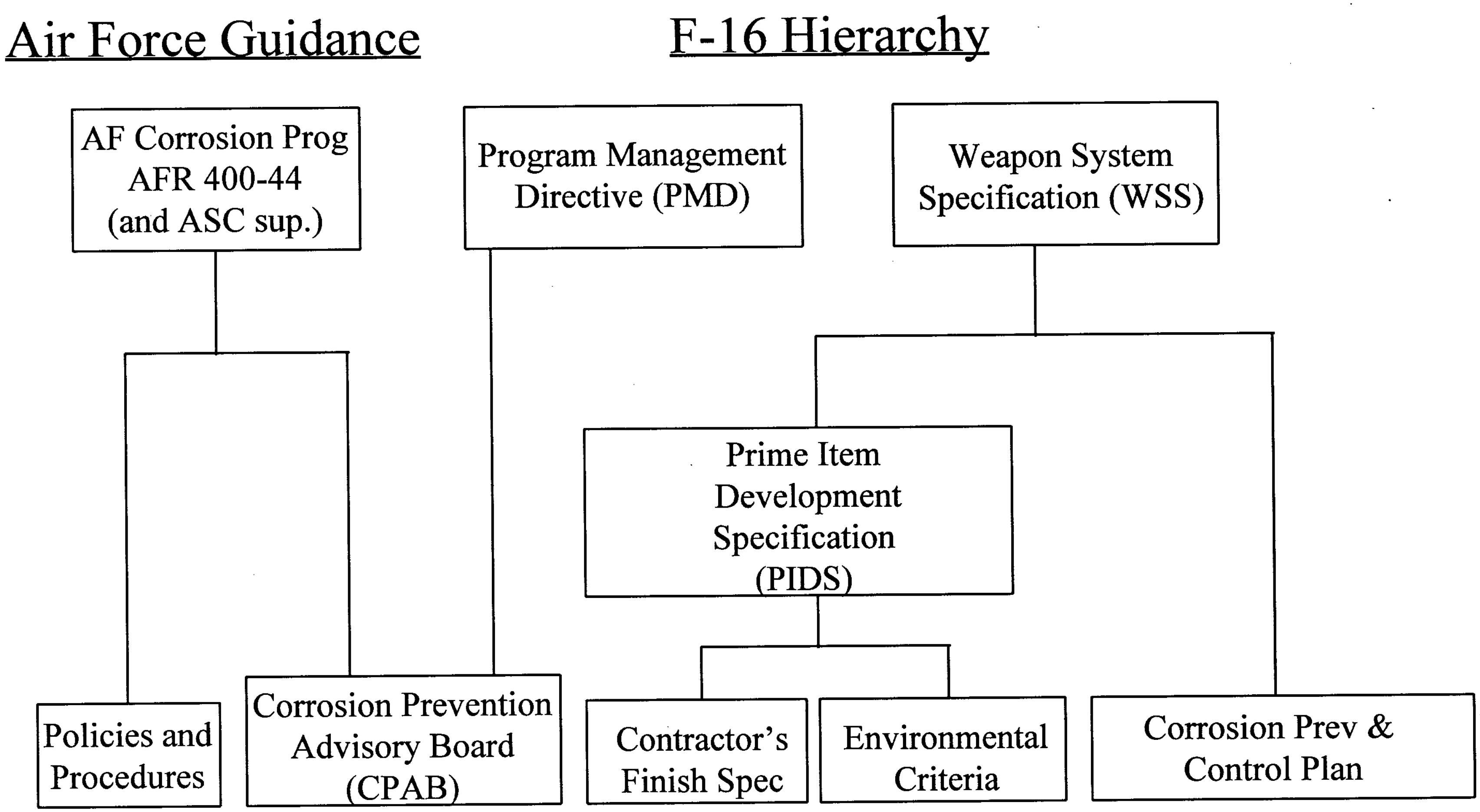


Figure 1 - F-16 CORROSION CONTROL IMPLEMENTATION

referred to as the CPAB, is aimed at the customer or user (field issues). A "B" meeting, the Corrosion Technology Interchange Meeting (CTIM), focuses on production issues and new technology. These meetings have been immensely helpful to the F-16, by keeping the lines of communication open between the field, contractor, labs, Air Logistic Centers (ALC), and SPO.

2. F-16 CORROSION PREVENTION SYSTEM

2.1 Primary Materials

To better understand the corrosion prevention system challenges, the material composition of the F-16 aircraft, as described in References [1-3], is detailed below. The majority of the F-16 aircraft is composed of aluminum (71%), steel (12%), titanium (1%), and composites (3%). Figures 2 and 3 show the general material usage on the airplane.

The fuselage skins and longerons are predominantly 2024 aluminum. The frames and bulkheads are primarily 2024 and 2124 aluminum, with some 7075 and 7475 aluminum as well. A titanium heat shield protects structure and hardware routing in the upper aft nacelle cavity over the hot burner section of the engine. The nose radome is a filament-wound fiberglass laminate.

Upper wing skins are 2024 aluminum, while lower wing skins are 7475 aluminum. The wing spars are mainly 7475 aluminum while most wing ribs are 7075 aluminum. The fastening system consists of cadmium-plated monel A286 CRES (Corrosion Resistant Steel) or alloy steel rivets. Wing-to-fuselage attachment fittings are 7475 aluminum. Wing hardpoints are PH13-8Mo CRES. The leading edge flap (LEF), flaperon, and fixed trailing edge flap have 2024 aluminum skins on full depth 5052 aluminum honeycomb cores, with some 2024 and 2124 aluminum substructure.

The vertical stabilizer is a bonded assembly of graphite-epoxy skins with 2024, 2124, 7049 aluminum, and 17-7PH CRES substructure. The fuselage attachment fittings are 7175 aluminum. The rudder has a 5052 aluminum honeycomb core with graphite-epoxy skins. Fiberglass plies cover the exterior and interior surfaces of the skin for protection against erosion of the outer surface and corrosion of the inner surface. (i.e., graphite and aluminum have a large galvanic potential).

The horizontal tail is a bonded assembly of graphite-epoxy skins with a corrugated 5052 aluminum under structure (ribspar). As on the rudder, fiberglass plies cover the exterior and interior surfaces of the skin. The pivot shaft and root rib are 6Al-4V beta-annealed titanium, while the outboard closure can be 2024 or A356 aluminum. Forward and aft wedges are fiberglass with 321 stainless steel caps.

The ventral fin is a bonded assembly of 5052 aluminum honeycomb core, 2024 aluminum skins, and 2124 aluminum ribs and attach fittings. The aft bay has a phenolic core (thermoplastic resin with glass fibers) and fiberglass skins.

The original F-16, through Block 30, had "lightweight" landing gear, while Blocks 40 and 50 have "heavyweight" landing gear. The change from lightweight to

Component		Aluminum	Steel	Titanium	Composites	Other*
PERCENT(%)		71	12	1	3	13
FUSELAGE	SKINS	2024				
	LONGERONS	2024				
	FRAMES/BULKHEADS	2024/2124				
	RADOME				FIBERGLASS	
WING	UPPER SKIN	2024				
	LOWER SKIN	7475				
	SPARS	7475				
	RIBS	7075				
	ATTACH FITTINGS	7475				
	HARDPOINTS		PH13-8MO			
	CONTROL SURFACES	2024/2124				
		5052honeycomb				
VERT STAB	SKINS				GRAPHITE-EPOXY	
	SUBSTRUCTURE	2024/2124	17-7PH			
		7049				
	ATTACH FITTINGS	7175				
	RUDDER	5052honeycomb			GRAPHITE-EPOXY	
HORIZ STAB	SKINS				GRAPHITE-EPOXY	
	SUBSTRUCTURE	5052ribspar				
	PIVOT SHAFT/ROOT RIB			6AL-4V beta-anld		
VENTRAL FIN	SKINS	2024			FIBERGLASS	
	SUBSTRUCTURE	2124/5052honeycomb				PHENOLIC
	ATTACH FITTINGS	2124				
LAND GEAR	LIGHTWEIGHT (PRE-BLK 40)	7049/7175	300M/4330V/D6AC			
	HEAVYWEIGHT (BLK 40/50)	7049	300M/4330V			
	WHEELS (HW & LW)		2014			

* "Other" pertains to copper, gasses, adhesive, insulation, sealant, wire, paint, fluids, rubber, plastic, etc.

Figure 2 - PRIMARY F-16 MATERIALS

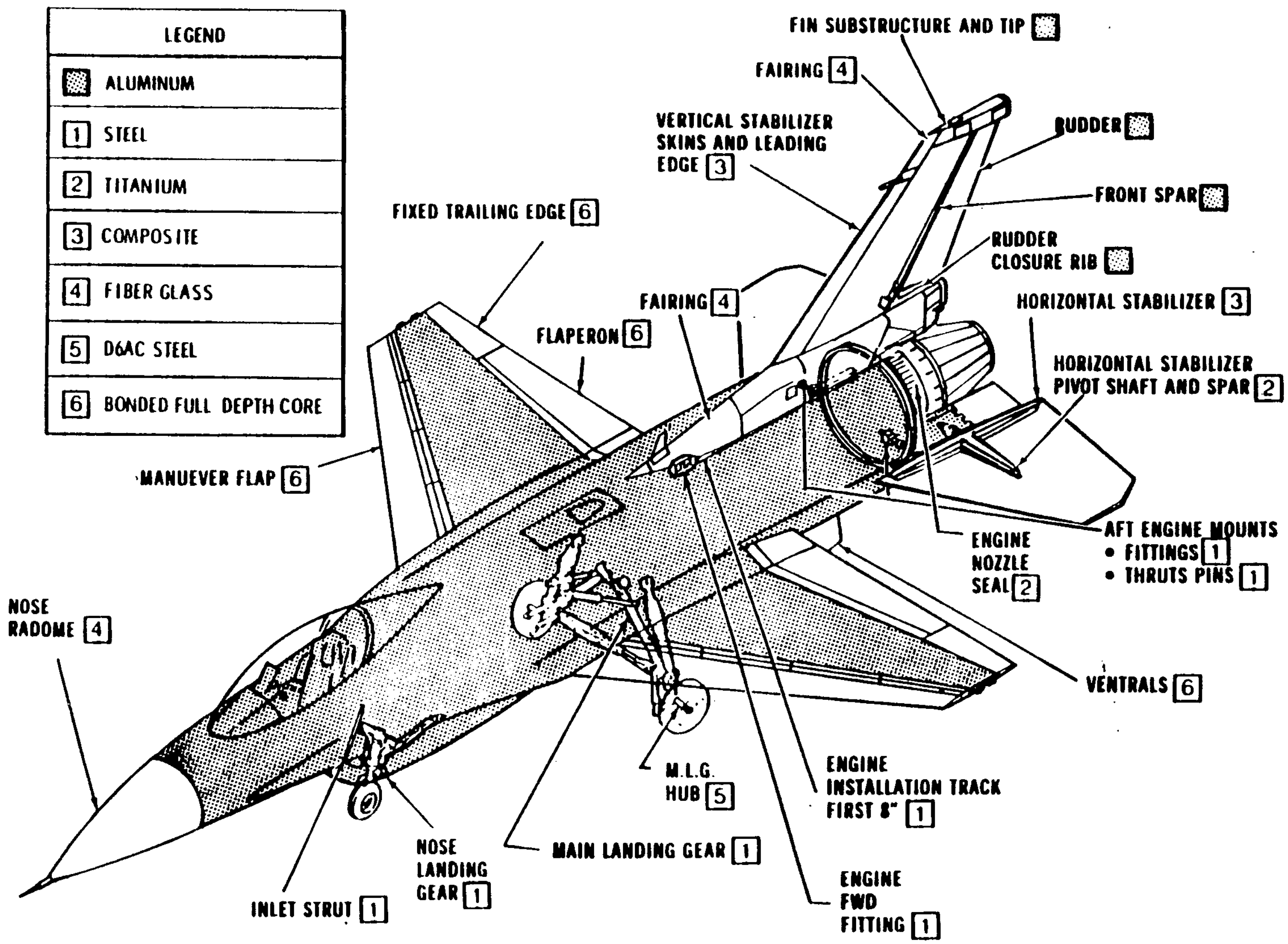


Figure 3 - PRIMARY F-16 MATERIALS

heavyweight gear was made due to the increased weight of the aircraft. Although both types of gear were constructed of a variety of materials, the lightweight gear had a fair number of aluminum components (7049 and 7175) as well as steel components (300M, 4330V, D6ac, etc.), while the heavyweight gear has very few aluminum parts. Wheels for both gear are 2014 aluminum.

2.2 Production Finish System

The F-16 finish system consists of cleaning, surface treatment, primers, and topcoats (figure 4). Cleaning to remove impurities is fundamental to achieving a surface that provides good adhesion for the paint system. Surface treatments serve two functions: 1) to improve adhesion of corrosion-inhibiting primer coatings and 2) to provide some corrosion protection by use of chromium in the process. A weather-resistant topcoat material is applied to the primer on the exterior and certain interior locations.

In general, most metal alloys are cleaned with mild abrasives, solvents, degreasers, or inhibited alkaline materials. Aluminum alloys may also use approved acid cleaners.

Surface treatment for 2000 and 7000 series aluminum alloys is primarily chromic acid anodize. Anodizing creates a protective aluminum oxide film on the surface of the part. The anodize process includes cleaning using vapor degreasing and oxide removal (or "descale/desmut") using triacid (hydrofluoric/chromic/nitric) etch, followed by the chromic acid anodize and a dichromate seal. Sulfuric acid anodize may be used as an alternative except on fatigue- or fracture-critical parts.

CRES and stainless steel, with some exceptions, are generally passivated. Passivation involves a mild acid bath that cleans off contaminants, (i.e., it removes free iron from the surface), that can bridge the inherent oxide layer of the metal. Non-CRES alloys and copper alloys are generally cadmium plated or receive aluminum ion vapor deposition (IVD). Chrome plating, electroless nickel plating, and tin plating are also used for specific applications on non-CRES alloys. Titanium and nickel alloys generally do not receive a surface treatment, while magnesium alloys receive anodic treatment. Stainless steel empennage leading edge caps are coated with a coating compound referred to as wash primer to promote adhesion of the primer to the base metal.

After cleaning and surface treatment, parts receive the primary corrosion barrier primer prior to assembly: either a low volatile organic compound (VOC) chromated waterborne epoxy primer, or a high VOC chromated epoxy polyamide primer. Before application of the topcoat, bare aluminum is touched up with a chromated chemical conversion coating. A mist coat of high-VOC chromated epoxy (lead-free) sealant primer is applied to promote adhesion to the bare portions of steel fasteners. A chromated high VOC flexible primer is then applied, which allows the paint system to flex without cracking, especially around fasteners. Finally, a polyurethane topcoat (either high or low

MATERIAL	ALUMINUM	STEEL	TITANIUM	OTHER
CLEAN	Mild abrasives, solvents, degreasers, inhibited alkaline cleaners. In some cases, acid cleaners.			
COMMON SURFACE TREATMENTS	Chromic acid anodize, or chromated chem film.	Cadmium plate or IVD aluminum. Also chrome, nickel, and tin plating. (CRES is passivated)	None or flouride phosphate treatment.	Anodic treatment for magnesium.
PRIMER	Low VOC chromated waterbourne epoxy, or high VOC chromated epoxy polyamide. Prior to assembly.			
SPECIALIZED PRIMERS	High VOC chromated epoxy sealant primer (lead free) to promote adhesion to bare steel, i.e. fastener heads, and high VOC chromated flexible primer to prevent cracking/peeling near joints and fasteners. Primarily used on exterior surfaces.			
TOPCOAT	High VOC polyurethane or low VOC polyurethane.			
SEALANTS	Polysulfide elastomer.			
RAIN EROSION	Leading edges - polyurethane tape or flexible primer and polyurethane rain erosion coating. Nose radome - antistatic fluoroelastomer coating.			

Note: This chart only shows the most common finish systems. There are many exceptions and variations.

Figure 4 - F-16 FINISH SYSTEM

VOC) is applied (figure 5). Other topcoats used include epoxy topcoat for equipment bays (i.e., better chemical resistance but poor ultraviolet resistance) and zinc chromate primer for some steel applications (i.e., bushings).

For rain erosion protection, the leading edges of the wing and empennage are covered with polyurethane rain erosion (RE) tape applied on top of the standard primer and topcoat. Until recently, the factory delivered aircraft without RE tape, and the user would install it after receiving the aircraft. An alternative rain erosion protection for the empennage uses flexible primer and polyurethane rain erosion resistant coating instead of the tape. The rain erosion coating on the nose radome is an antistatic fluoroelastomer.

Fuselage fuel areas use a heat-cured corrosion protective coating applied to integral fuel tank interiors during manufacture.

The primary sealants on the F-16 are polysulfide elastomers. These sealants are widely used for fastener installation and faying surfaces, as well as aerodynamic smoothing and form-in-place gaskets. Chromated sealants were used in recent years for fastener installation but are currently being phased out in an effort to reduce chromium use.

2.3 Field Finish System

Historically, there has not been a significant difference between the field/depot and production finish systems applied to the aircraft. The field

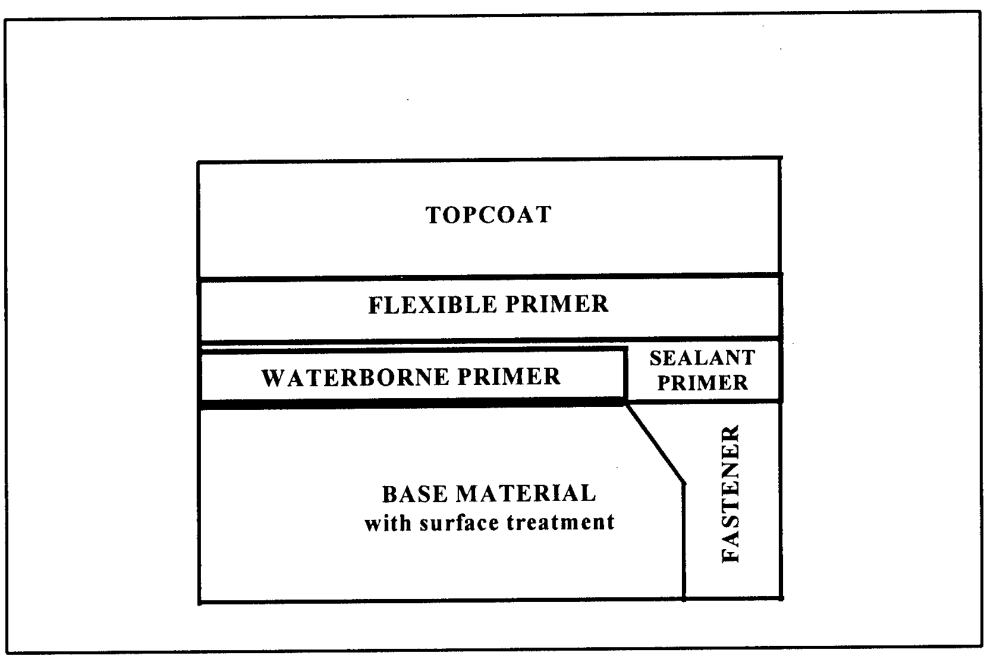


Figure 5 - F-16 FINISH SYSTEM

procedure is limited to touch-up and scuff sand/repaint, while the depot can also perform a total strip/repaint using plastic media blasting (PMB). One of the primary differences between the field/depot and production finish systems has been a more extensive use of chemical conversion coating to touch up aluminum when anodize is missing. There is not much difference between field/depot finish system and the production system.

In 1990, the USAF began using environmentally compliant paint systems. The F-16 program began using new equipment (HVLP spray guns) along with new waterborne primers and high solids (low VOC) paints. Because there were significant problems with the primers and topcoats (e.g., mixing, application), the F-16 depot began a test program in 1993 to develop a new environmentally compliant paint system for the F-16. The study has concluded that, for total repaints, the preferred method involves a PMB strip followed by an aircraft wash and solvent wipe to clean the PMB residue from the surface. A light sanding combined with a corrosion remover promotes adhesion for chemical conversion coating. A low-VOC chromated flexible epoxy primer and low-VOC polyurethane topcoat provide the needed environmental barrier. The field process for total repaints is identical to the depot process but starts with a scuff sanding rather than PMB.

Wright Laboratory is also studying paint systems for the "now-term," "mid-term," and "long-term." The now-term study evaluates off-the-shelf paints and primers for use by the ALC's and field units. The evaluation includes screening tests, full laboratory tests, and in-service testing. Screening tests primarily address adhesion and aesthetics, while lab tests assess corrosion protection characteristics. In-service testing is done for two paint systems in three environments (cold/hot/wet) on several aircraft systems. The mid-term and long-term studies will alter coating formulations to try to achieve zero VOC and no chromium, and a life equivalent to that of the aircraft system.

2.4 Cleaners and Wash Program

Cleaning and washing are not only fundamental to maintaining the appearance of the finish, but are important aspects to the overall corrosion prevention program. This is because cleaning and washing prevent contaminants from damaging the finish, remove electrolytes conducive to corrosion, and remove soils, which can hold moisture against the structure.

Depending on geographic location, aircraft wash and rinse intervals are specified in the Aircraft Weapons Systems Cleaning and Corrosion Control technical order. Obviously, locations near salt water, high humidity, and/or heavy industrial pollution have shorter wash and rinse intervals than other locations. The corrosion survey findings section of this paper explains how frequent rinsing plays a large role in corrosion prevention and finish preservation. Approved cleaners include turpene based, alkaline, solvent emulsion, and other types.

3. PAST PROBLEMS AND SOLUTIONS (LESSONS LEARNED)

Canopy frames (figure 6): Dissimilar metals and water entrapment have caused corrosion in canopy frames. The dissimilar metals are the aluminum canopy frame that makes contact with a silver bus bar. This bar acts as an electrical ground to prevent build-up of static charge on the transparency. A separate problem occurs when moisture becomes trapped within the canopy frame. Solutions have included issuing repair guidelines, providing drain holes, and replacing silver with tin in new designs. Replacement of conventional transparency sealants with "dryseal" (similar to automotive doors and windows) in new designs and for preferred spares will also help the moisture intrusion problem. This substitution, however, was driven not by moisture problems but by the polycarbonate transparency crazing problems caused by the polysulfide sealant.

Leading edge flap (LEF) outboard hinge pin (figure 7a). Original hinge pins were cadmium-plated H-11 steel. Because of the inboard-outboard sliding motion of the LEFs, the cadmium plating wore away and corrosion occurred. The corrosion seized the pin in the bushing, resulting in fatigue cracking of the hinge fitting, which was not designed to react inboard-outboard loads. A TCTO was issued to install more durable chromium plated H-11 steel pins.

Leading edge flap (LEF) torque tubes (figures 7a and 7b): Early LEF torque tubes, made of cadmium plated 4130 steel, did not experience corrosion problems. Because of wear concerns, the tubes were redesigned and foam and elastomer plugs were added inside the tubes near the splined ends of the shaft to prevent grease from migrating away from the ends. Severe corrosion resulted due to the combination of a poor moisture seal, the water retention characteristics of the foam and rubber, and inadequate or missing cadmium plating on the tube inside diameter. The problem was discovered when a torque tube failed in flight in October of 1994. Upon further investigation, a large lot of these tubes was found to have inadequate or missing cadmium plating. The solution identified was to perform visual inspections on aircraft with over 400 flight hours via a routine 90 day TCTO. In addition, an NDI of all affected aircraft was ordered at the next phase inspection interval, and new torque tubes were installed where required.

A second redesign for a preferred spare is in work, which consists of a 2024 aluminum torque shaft and 4340M steel couplings. This design was in work prior to the discovery of the corrosion problem; therefore, it focused primarily on improving the wear characteristics of the shafts. However, the potential for dissimilar metal corrosion has been addressed in the redesign: The aluminum tube is anodized, and the surface that mates with the steel coupling is coated with zinc chromate primer. The coupling has a titanium-cadmium finish and is mated with the aluminum tube in a magnaforming process. Testing to-date has included an 860-hour flight test followed by laboratory evaluation. The evaluation

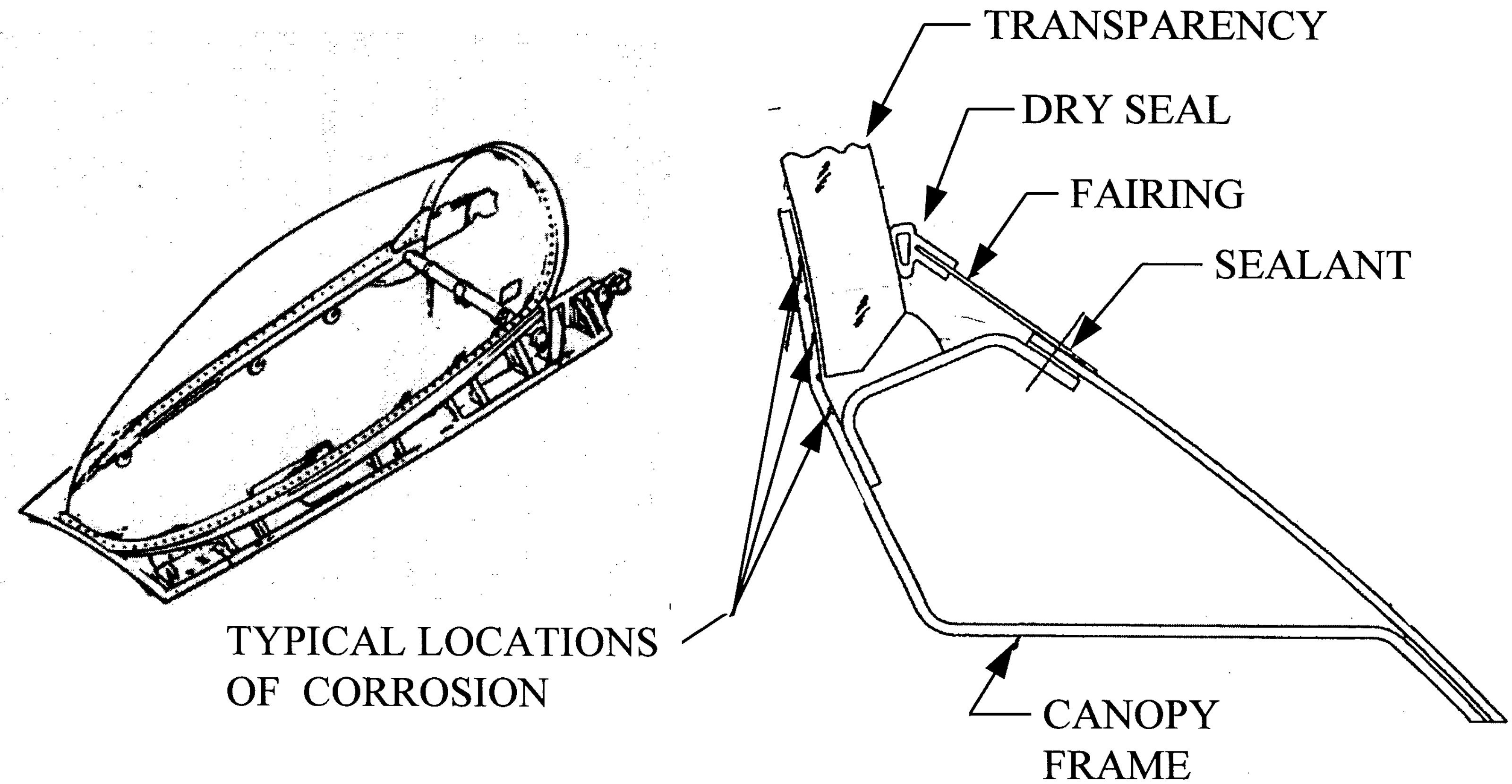


Figure 6 - CANOPY FRAME

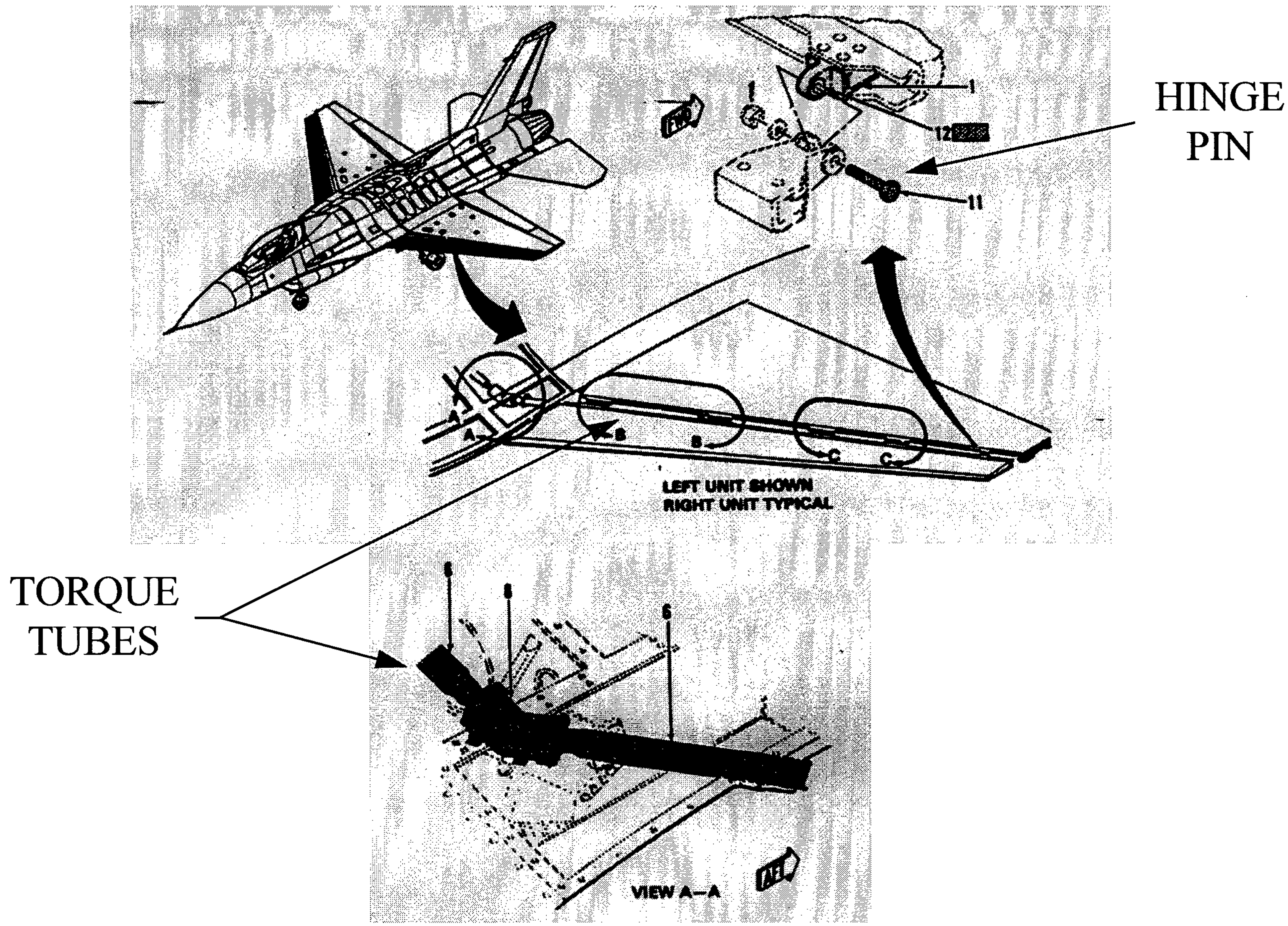
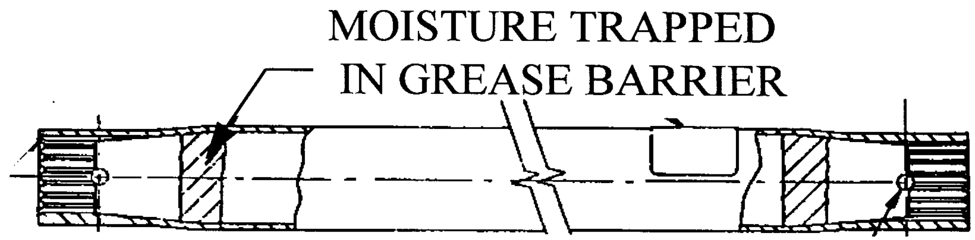
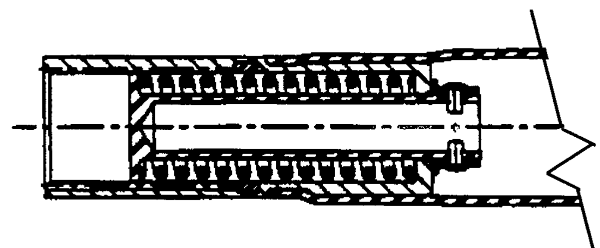


Figure 7a - LEF HINGE PIN AND TORQUE TUBES



ORIGINAL TUBE



REDESIGNED TUBE

Figure 7b - LEF TORQUE TUBE

recommended an epoxy primer with strontium chromate instead of the zinc chromate primer, and additional moisture protection using polysulfide sealant. These changes were implemented and further corrosion testing is being performed per the torque tube specification, expected to be complete in mid 1996. The new tube should be available to the field in late 1996.

Lightweight (pre-block 40) main landing gear (MLG) tension struts (figure 8). The lower collar lugs (axle housing end) of the tension strut, made of cadmium plated 300M steel, failed due to stress corrosion cracking. An investigation found that the cause was improper fillet sealing of the bushing at the bushing flange/collar flat interface. Cadmium plating becomes damaged when the bushing is installed, and poor sealing allows exposure to the environment. Corrosion pitting was followed by stress corrosion cracking. Depot sealing procedures were found to be adequate; however, the manufacturer's sealing procedures were suspect. The solution identified was to correct the manufacturer's process and conduct inspections for fielded gear including corrosion removal and resealing of the bushing/strut interface as required.

Main landing gear (MLG) wheel bolts. A European customer reported MLG wheel bolt failures caused by stress corrosion on F-16 A/B aircraft. Shortly afterwards, the customer determined that maintenance personnel were using an unauthorized cleaner, which actually removed the protective cadmium plating from the bolts. This is a good example of why it is important to refrain from using unauthorized cleaners.

Wing rib/hardpoint interface (figure 9). A European customer noticed during maintenance that white powdery deposits had formed near the wing hardpoint for the 370 gallon fuel tank pylon's forward attachment. Analysis of the powder showed that it was aluminum oxide, indicating possible corrosion inside the rib. The insert was removed and corrosion was found on the inside surface of the 2124 aluminum wing rib, which holds the PH13-8Mo CRES fitting. Corrosion damage was severe enough to prohibit carriage of tanks on several aircraft based on preliminary damage limits. An initial investigation found that the wing rib was not properly treated (anodize, prime, seal), allowing a path for the environment to enter into the exposed aluminum. It was determined that a production process omitted proper treatment on a large number of aircraft at some time in the past due to unclear procedures. Current production procedures were clarified and are believed to be adequate. However, further investigation showed that seal damage occurs during service even in properly manufactured assemblies. As a result, all aircraft will be inspected and repaired via TCTO at the next phase interval, and subsequent inspections will follow.

4. SUMMARY OF 1995 F-16 FIELD CORROSION SURVEY

4.1 Overview

Several members of the F-16 Corrosion Prevention Advisory Board formed a

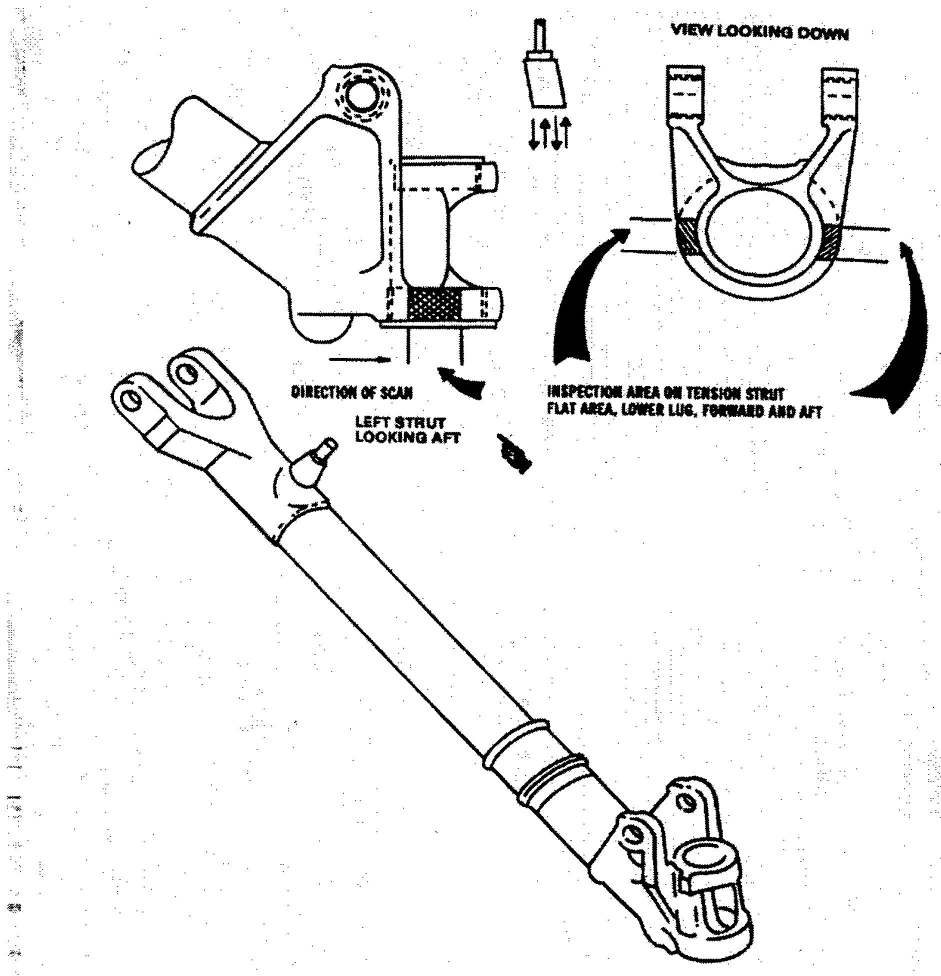
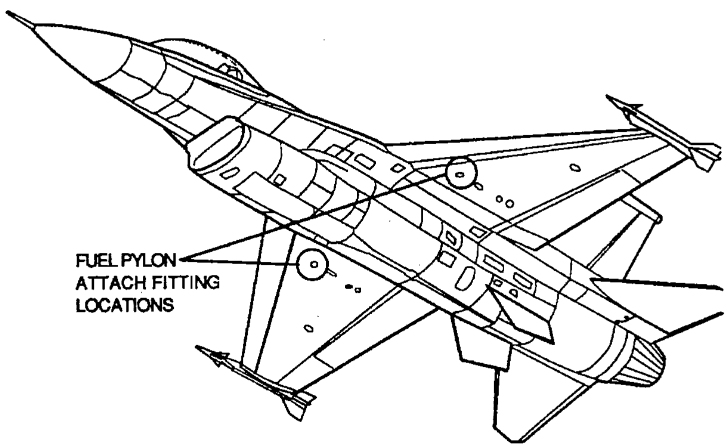


Figure 8 - MAIN LANDING GEAR TENSION STRUT



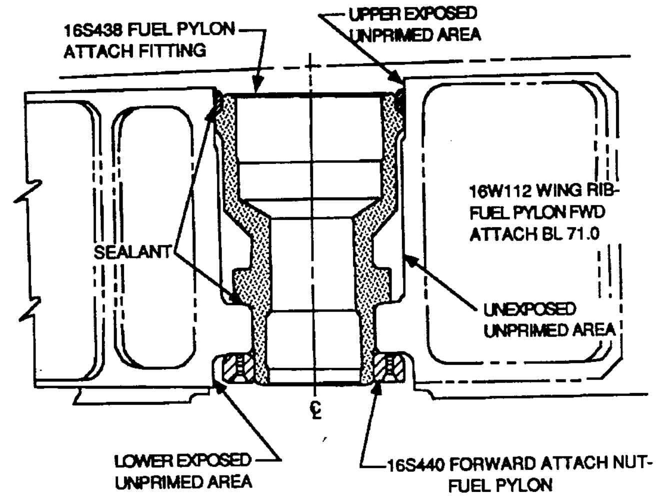


Figure 9 - WING RIB

team and conducted a field survey from 1-31 May 1995. The intent of the survey was to assess the overall condition of the F-16 corrosion prevention system, including aircraft, F-16 peculiar subsystems, corrosion control facilities, phase hangar, backshops, equipment, procedures, and practices. This was accomplished by visiting nine (9) units worldwide that represented a large population of F-16 aircraft, users, and environments.

4.2 General Observations

Unauthorized cleaners and materials are routinely used, especially in the USAF. One extreme case of unauthorized cleaners outside the USAF resulted in stress-corrosion cracking of main landing gear wheel bolts, mentioned earlier. Also, although no severe problems have resulted to-date, an unauthorized paint commonly known as "white-out" (i.e., typewriter correction fluid) was used to touch up white paint in landing gear areas. White-out composition has not been evaluated, but this is a sensitive area with high strength steel parts susceptible to hydrogen embrittlement and stress corrosion cracking.

Although electrical connectors with nickel (Ni) plating have been superseded by nickel-cadmium (Ni-cad) plating, many Ni connectors are still in service and corrode heavily if corrosion preventative compounds (CPC) are not used. Also, CPC's are not permanent coatings and need to be reapplied over time.

Outdoor storage causes accelerated aging of the finish system and is a problem for most users. The problem is compounded at units that routinely park aircraft on the flightline with open canopies. Some FMS (Foreign Military Sales) customers have shelters for all aircraft and related subsystems.

Aggressive wash and rinse programs proved to be one of the most effective means of maintaining the finish system and preventing corrosion. A good example was found at a USAF base in which one particular crew chief rinsed his aircraft two to three times as often as required. Other aircraft that had identical finish systems (same age, same materials), had become faded and chalked while the finish of the frequently rinsed aircraft was in excellent condition.

Paint adhesion was a problem at several units when surface preparation or specialized primers were omitted from the finish system (figure 10). In most cases, small areas had peeled and most fasteners were exposed and rusted. The more common omissions included lack of chromated chemical conversion coating on bare aluminum, lack of chromated epoxy (lead free) sealant

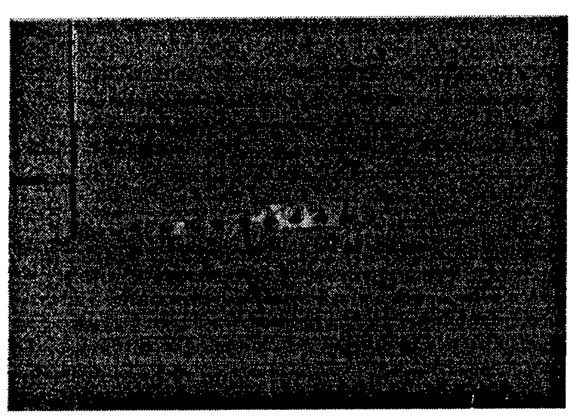


Figure 10 - Inadequate Paint Adhesion

primer on bare steel (i.e., fastener heads), and lack of flexible primer near fasteners.

4.3 Subsystem Corrosion

Some of the more significant subsystems found to suffer from corrosion include the F110 engine accessory gearbox main housing and LANTIRN targeting pod.

The gearbox housing is anodized magnesium coated with a silicone enamel topcoat and is located beneath the engine. An in-flight emergency, due to loss of oil pressure, resulted in the discovery of a through hole in a recessed pocket of the housing. The recessed pocket collected moisture, which resulted in corrosion. An immediate action TCTO was issued to visually inspect the recessed pocket with a borescope, and if necessary, perform an interim repair. The interim repair consists of an isopropyl alcohol wipe, primer and topcoat. Damage limits are very restrictive and no material removal is allowed due to the critical web thickness of the pocket. A permanent repair is still being developed. Production processes have changed to add a resin seal following the anodize process. This improves corrosion protection as well as adhesion characteristics (figure 11). Wright Laboratory has recommended the following permanent field repair process: 1) incorporate mechanical abrasion to remove corrosion, 2) apply corrosion inhibiting primer, 3) fill the recessed area with polysulfide sealant, and 4) topcoat with a gloss polyurethane. Redesign using a new material and/or geometry is not being pursued.
Nearly all LANTIRN targeting pods were experiencing severe paint delamination on the forward shroud of the Nose Equipment Support Assembly (NESA) (figure 12). Information was passed to the LANTIRN SPO and a subsequent investigation determined that the finish was originally a black nickel but was eliminated in 1988 and replaced with black anodize. The paint process was changed again in 1994 from black anodize to a conversion coat or chem film chromate. All the finishes require some surface preparation to promote adhesion for subsequent coatings. No production changes are planned, but the field was alerted of the situation. Also, the current T.O.'s already require a scuff sanding, prime, and topcoat; these procedures apparently were not being performed.

4.4 Action Items Generated

Some significant items resulting from the survey include the following:

Depot now offers rain erosion (RE) tape installation on aircraft leading edges as an option for total aircraft repaints. Field units have complained of paint peeling on leading edges of wings and empennage of newly delivered aircraft and on existing aircraft following total repaints. Production aircraft were recently changed to install tape prior to delivery to the user, but the depot continued to deliver repainted aircraft without tape. In extreme cases, leading edge damage would occur during the flight from the depot to the field unit. Also, updated technical orders permit use of RE tape on several other leading edges (e.g., stores, pylons).

Studies are underway to develop improved methods to strip spray-on rain

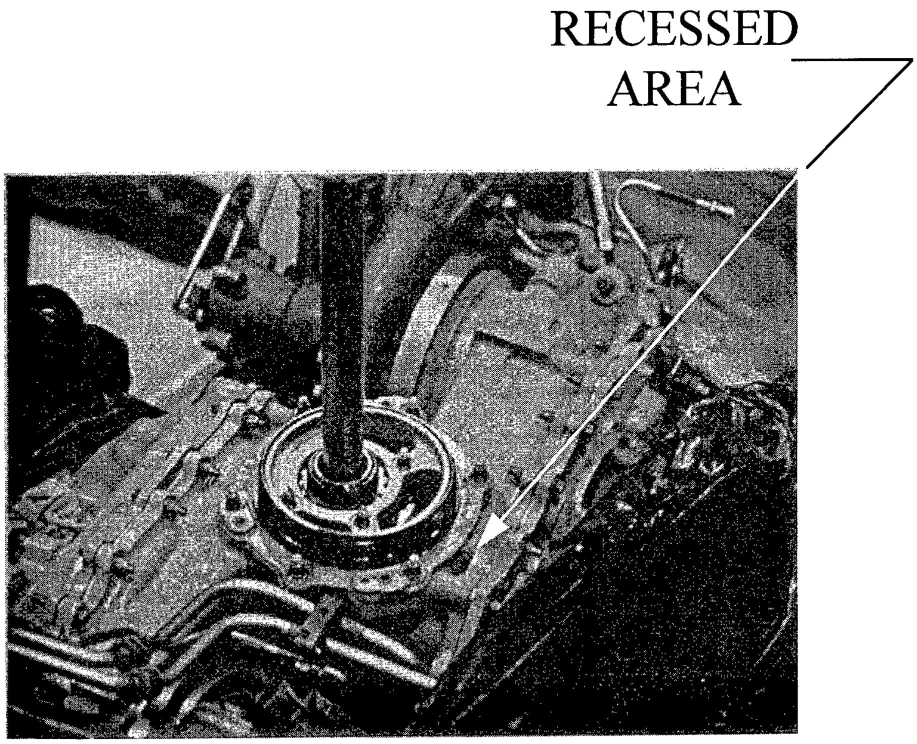


Figure 11 - ENGINE GEARBOX

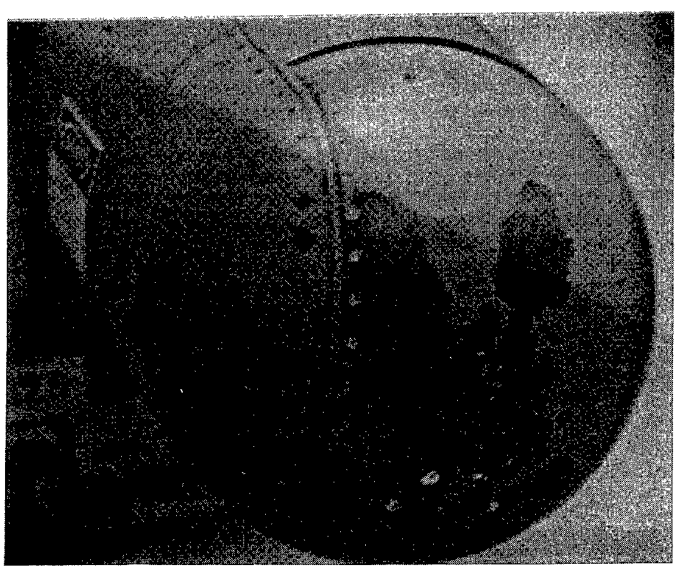


Figure 12 - LANTIRN POD

erosion coatings. Thousands of aircraft still have spray-applied RE coatings on composite empennage leading edges, and environmentally compliant removal methods for repairs or change-over to RE tape is time consuming and labor intensive. The studies are evaluating both mechanical and chemical means of removal.

Several miscellaneous items have experienced severe corrosion due to poor design. Most of these items are planned to be improved by a material change, a finish change, or better sealing. A typical example is the inflight refueling receptacle scuff plate and attachment fasteners, which are cadmium-plated steel. Repeated impacts from refueling equipment damage the cadmium plating and expose the steel, which rusts quickly. A material change to CRES should eliminate the problem.

Several years ago, the spring retainer material for the nose landing gear (NLG) downlock actuator was changed from non-CRES to CRES to prevent corrosion which could seize the retainer in the NLG downlock actuator (figure 13). The T.O.'s cited the inspection and preferred spare but it was not enforced via a TCTO. This became evident at one of the bases visited in which all aircraft had the old material and exhibited severe surface corrosion.

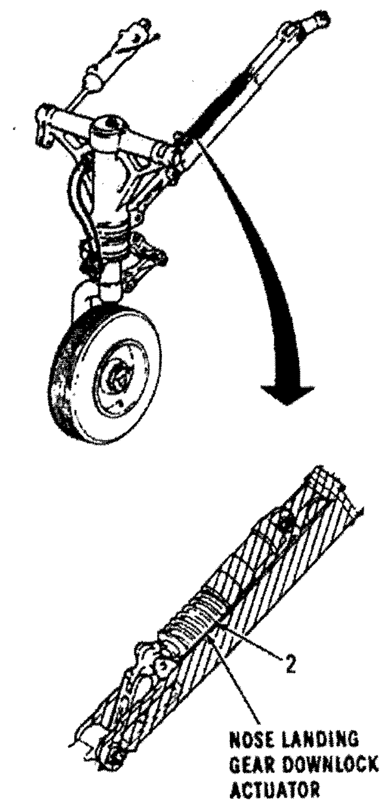
Other items that have experienced corrosion include the aluminum NLG tension strut collar (figure 14), and steel inner cylinder. These normally are not inspected but were detected during a routine teardown by one of the European customers. Corrosion was severe, and the parts required replacement. No other user has ever reported similar findings, but, as mentioned, this is not a scheduled inspection. Plans are to include all of these items, along with several other non-corrosion-related items, in an upcoming NLG teardown inspection TCTO.

Limited testing is being conducted to find more durable replacements for dry film or solid film lubricants (SFL) used in the field. The field uses an air dry SFL with a molybdenum disulfide (MoS_2) lubricant. Production uses a heat-cured lubricant. SFL is used on several components, not only for lubrication, but for corrosion protection as well. Unfortunately, the SFL does not last long and is a maintenance burden. Some items that use SFL include canopy hinges, gun barrel and related components, and bomb rack components (e.g., solenoids, threaded studs, etc.).

In general, it was found that although the F-16 is a mature system, there is still room for improvement. The survey reinforced the fact that attention to corrosion prevention can never be relaxed since corrosion is time-dependent, and new problems can surface anytime during the life of the system.

5. ONGOING AND FUTURE EFFORTS

Sidestick controller (SSC) (figures 15a and 15b). The SSC is a limited displacement force sensor that the pilot uses to provide pitch and roll commands to the aircraft's flight control system. The SSC contains a diaphragm, a 0.008



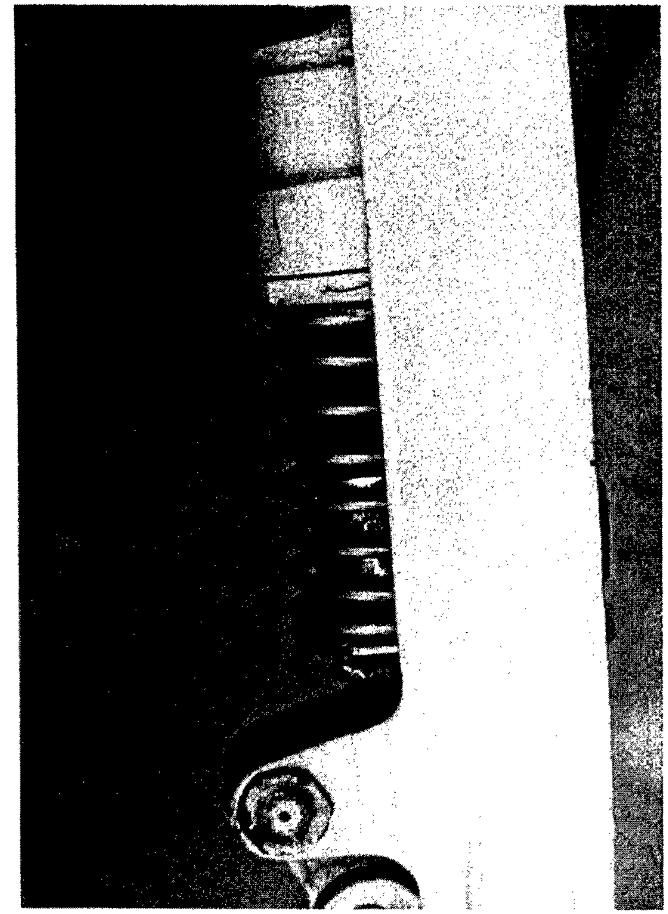


Figure 13 - NOSE LANDING GEAR DOWNLOCK ACTUATOR

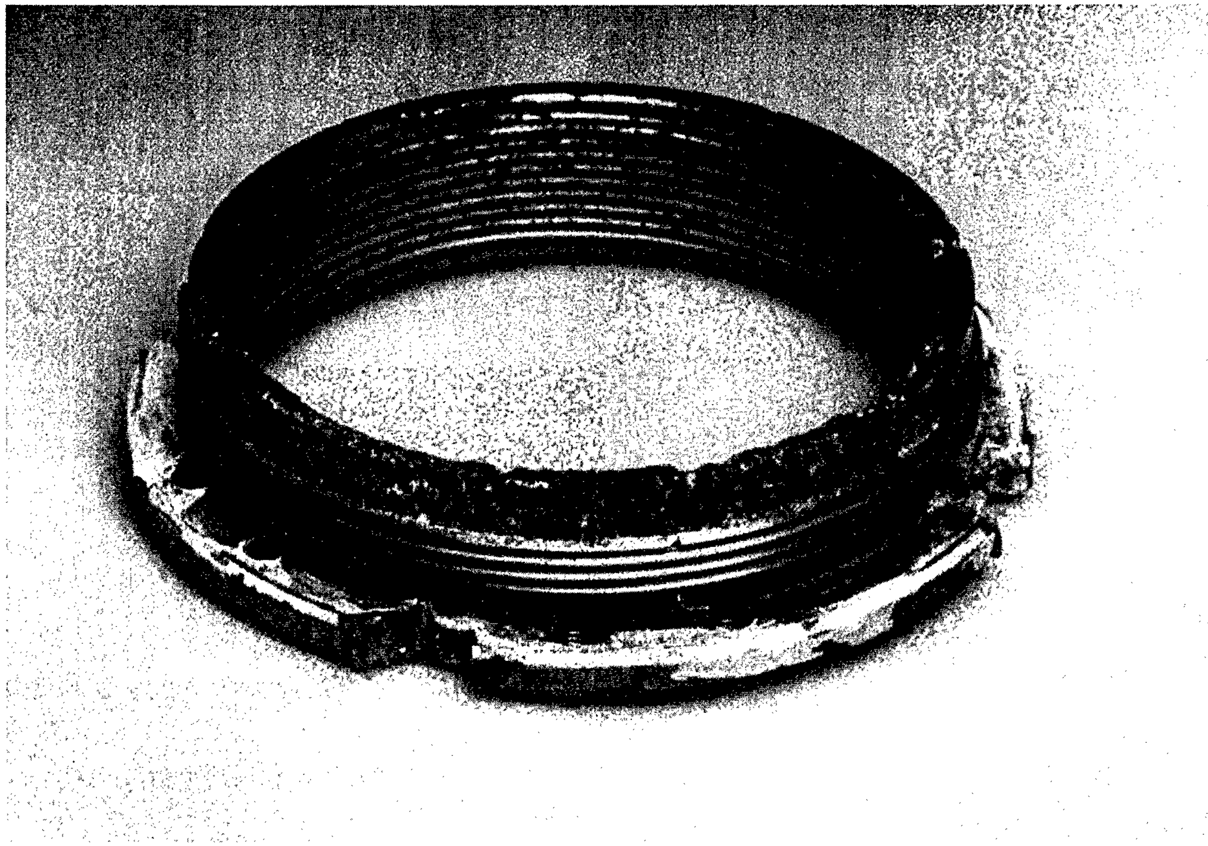


Figure 14 - NOSE LANDING GEAR TENSION STRUT COLLAR

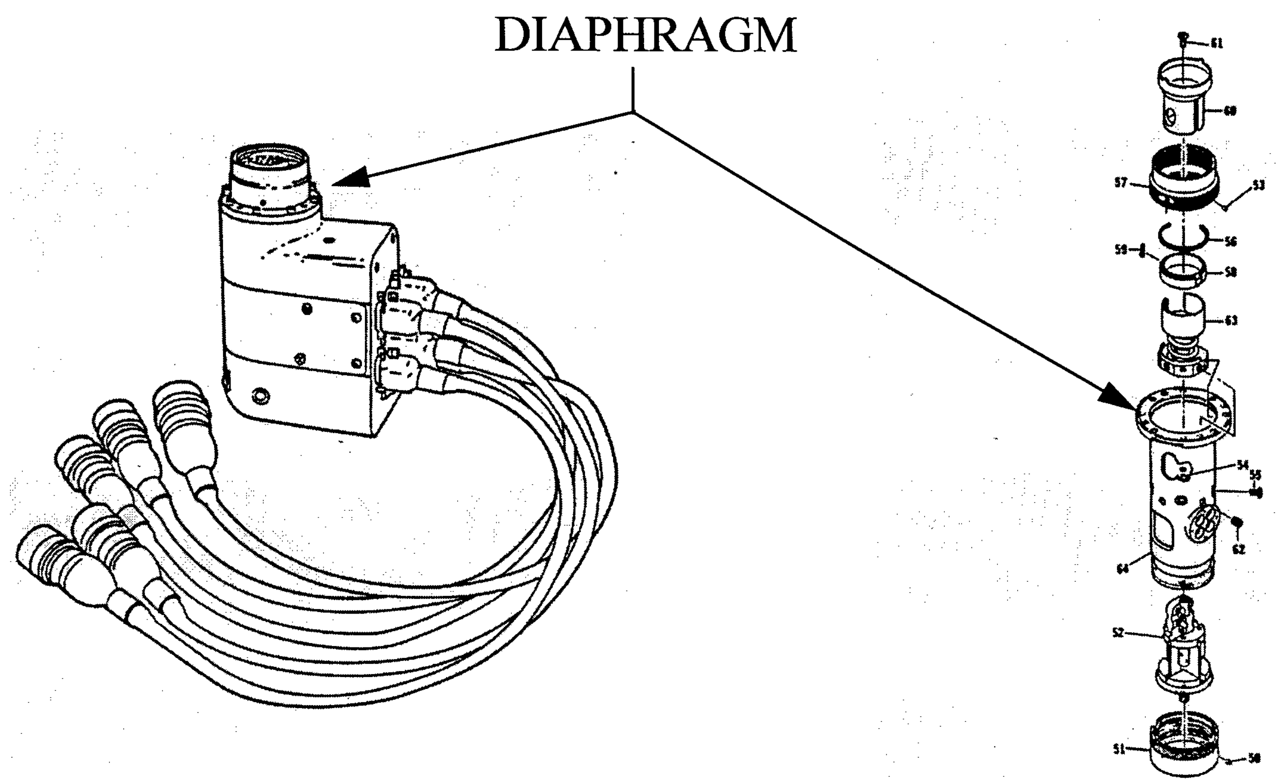


Figure 15a - SIDESTICK CONTROLLER

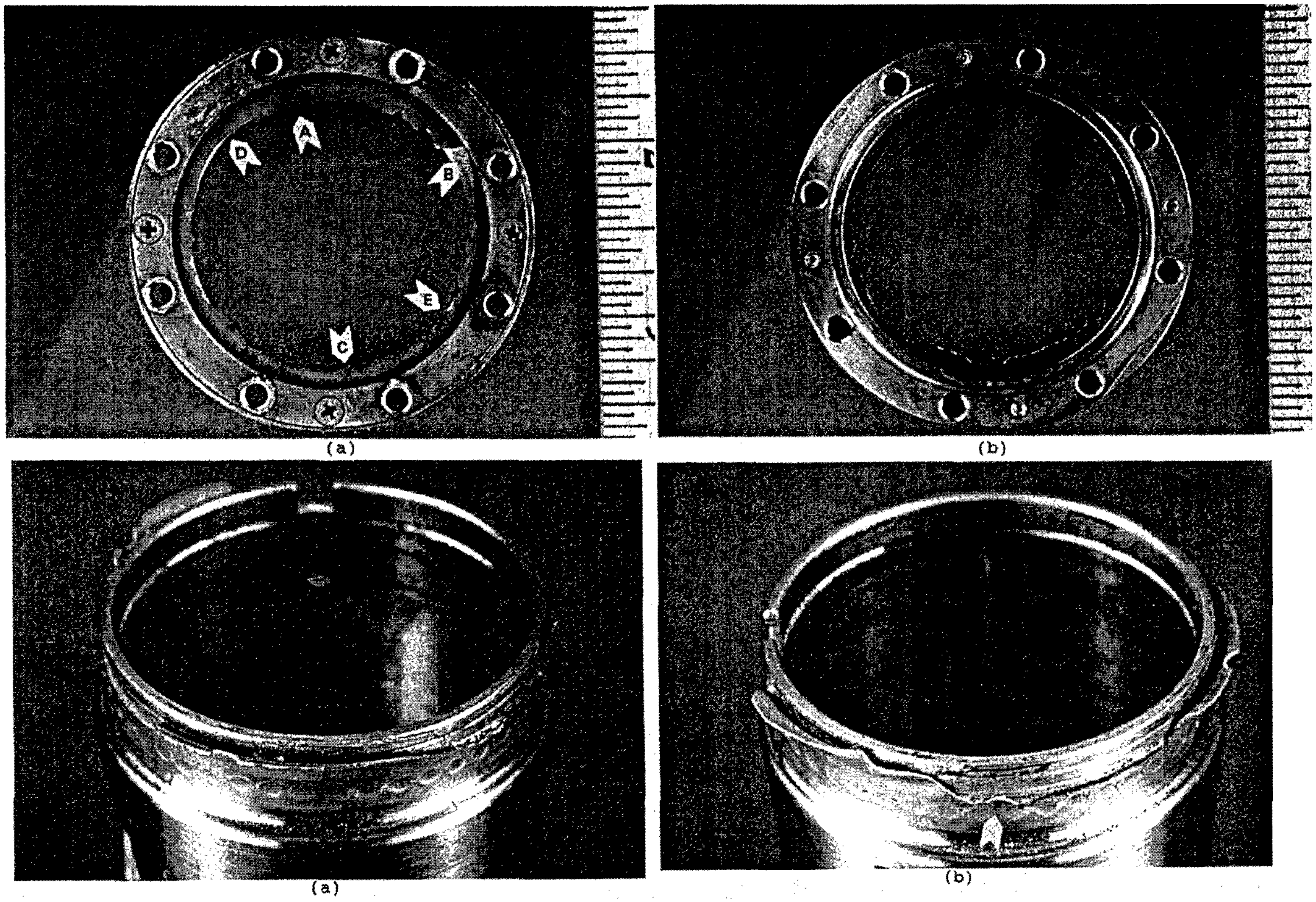


Figure 15b - SIDESTICK CONTROLLER DIAPHRAGM inch thick 17-7PH (RH950) CRES component, which receives the forces applied to the SSC and flexes in response to those forces. An in-flight failure of a sidestick controller diaphragm in early 1995, fortunately occurring in a two-seat F-16, prompted an investigation. The investigation found a similar failure occurred approximately 10 years prior, resulting in the requirement to coat the diaphragm with a CPC (MIL-C-85054, Amlguard). The intent was to routinely inspect and reapply the CPC, but the inspection was never added to the T.O.'s. The breakdown of the CPC resulted in intergranular attack indicative of stress corrosion cracking. A massive inspection was conducted to remove any sidesticks that showed signs of corrosive attack, and replace them with sidesticks that have received the CPC application in production that have not experienced problems to date. Continued inspections are being considered, but concerns regarding inspection techniques and reapplication of CPC's have put further inspections on hold. Studies are underway to find improved materials and coatings, and also to consider new features such as a protective boot and/or mechanical backup.

Environmental Protection Agency (EPA-17) Reductions. The Air Force has been directed, by USAF policy 94M-003, to reduce as near to zero as possible the number of EPA-17 chemicals (figure 16) being used by the Air Force. The United States EPA has also passed legislation restricting our ability to use these chemicals. In order to comply, costly modifications are required in the production and maintenance lines. Several projects are planned to pursue compliance with current legislation by looking at source reduction efforts as opposed to the costly control technologies. The projects include studies of non-chromated sealants and primers, EPA-17 solvent reduction/elimination, cadmium dust exposure reduction, low VOC coatings, and environmentally friendly depainting technology. In addition miscellaneous chromium reduction or elimination efforts are being pursued, including replacement of chromic acid anodize with a thin film sulfuric acid anodize, and powder coating alternatives to VOC coatings.

Benzene	Methyl Ethyl Ketone (MEK)
Cadmium (and compounds)	Methyl Isobutyl Ketone (MIBK)
Carbon Tatrachloride	Nickel (and compounds)
Chloroform	Tetrachloroethylene (PERC)
Chromium (and compounds)	Toluene
Cyanide (and compounds)	1,1,1 Trochloroethane (TCA)
Lead (and compounds)	Trichloroethylene
Mercury (and compounds)	Xylenes
Methylene Chloride	

Figure 16 - EPA 17

6. ACKNOWLEDGMENTS

The author would like to extend special thanks to the following individuals for contributing their invaluable experience and expertise regarding the F-16 aircraft corrosion prevention system:

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Mark Carroll, Jr. (Senior Materials & Processes Engineer, Lockheed Martin Tactical Aircraft Systems).

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VISUAL AND NON-DESTRUCTIVE INSPECTION TECHNOLOGIES

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1.0 INTRODUCTION

Military and commercial aircraft are being used well beyond their projected service life with the result that the number of aging aircraft in the inventory is increasing. Also, the performance requirements of many military aircraft have increased compared to their original design. Maintaining the airworthiness of these aircraft is of prime concern to the regulatory authorities. One technology area which plays an important role in assuring the safety of flight of these aircraft is the proper inspection at regular intervals. Reliable visual and nondestructive inspection (NDI) methods are needed to assure the airworthiness of these aircraft and at the same time keep maintenance costs to a minimum.

This paper discusses currently available techniques for detecting damage in structures and their limitations. Inspection of cracks in substructure and hidden corrosion has always presented a nightmare for NDI engineers. Some recent advances made in the NDI technology to solve these problems are discussed.

2.0 COMPARISON OF NDI METHODS

A number of visual and nondestructive inspection methods are available for inspection. However, their application to detect flaws depends on the type of structure, access, desired degree of accuracy and inspection time. The comparison of conventional NDI methods is shown in Figure 1.

NDI Method	Ultrasonic	Eddy Current	Radiography	Penetrants	Magnetic Particle
Flaw Type	All	Cracks, Corrosion	All Except Small Cracks	All	All
Sub-surface	All	Shallow	All	Surface only	Shallow
Area of Scan	Small	Small	Large	Large	Medium
Flaw Sizing	Fair	Poor	Good	Very Good	Good
Test Time	Slow	Slow	Very Slow	Varies	Fast

Figure 1. Comparison of NDI Methods

Paper presented at the AGARD SMP Lecture Series on "Aging Combat Aircraft Fleets — Long Term Applications", held in Madrid, Spain from 7-8 October 1996; Pomezia, Italy from 10-11 October 1996; Atlanta, USA from 4-5 November 1996 and Brussels, Belgium from 22-24 January 1997, and published in LS-206.

The advantages and disadvantages of various NDI methods (Reference 1-2) are shown in Figure 2 along with their applications. Some of these techniques are discussed in the following paragraphs.

NDI Method	Detection Application	Advantages	Disadvantages
Visual	Large Surface Defects or Damage in all Materials	Simple to use	Reliability depends on experience of user
Optical	Surface defects/structural damage in all materials	Rapid large area inspection Good for bonded and cored structures	Accessibility required for direct visibility
Penetrant	Surface cracks in metals	Simple to use, accurate, fast, easy to interpret	Surface defects only, access required, defect may be covered
High Frequency Eddy Current	Surface defects, cracks, intergranular corrosion, pits, heat treat	Useful for detecting cracks at holes not detectable by visual or penetrant, fast, sensitive, portable	Trained operators, special probes for each application, reference standards required
Low Frequency Eddy Current	Subsurface defects, corrosion thinning	Useful for detecting cracks under fasteners or substructure without disassembly	Trained operator, time consuming, special probe for each application
Sonic	Delaminations, debonds, voids, and crushed core in composites, honeycombs	One side access, does not require paint removal or surface preparation	Difficult to interpret results, loses sensitivity with increasing thickness
X-Ray	Internal flaws and defects, corrosion, inclusions and thickness variations	Eliminates disassembly requirements, permanent record, high sensitivity	Radiation hazard, trained operators, crack plane must be parallel to x-ray beam, special equipment
Magnetic Particle	Surface and sub-surface defects in ferromagnetic materials	Simple, portable, easy to use, fast	Trained operator, parts to be cleaned before and demagnetized after check Magnetic flux must be normal to defect plane
Ultrasonic	Surface and sub-surface defects, cracks, disbonds in metals and composites	Fast, easy to operate, accurate, portable	Trained operator, test standards required, electrical source needed

Figure 2. Relative Advantages and Disadvantages of NDI Techniques

3.0 PROBABILITY OF DETECTION (POD)

Probability of detection (POD) is a statistically based quantitative measure of inspection capability. The POD is different for different inspection equipment and even for the same NDI equipment is affected by a number of factors such as: material properties, structural details, defect shape, inspection conditions, etc. Another parameter generally associated with POD is the confidence level with which a flaw can be detected. A 95% confidence level is considered acceptable for flaw detection. An NDI equipment capability is generally designated as 90% probability of detecting a flaw with 95% confidence level. The POD of various NDI equipment for through the thickness damage (Ref. 3) is shown in Figure 3. Figure 4 shows POD for sub-surface and internal defects. These figures indicate that the probability of detection varies significantly with each NDI equipment.

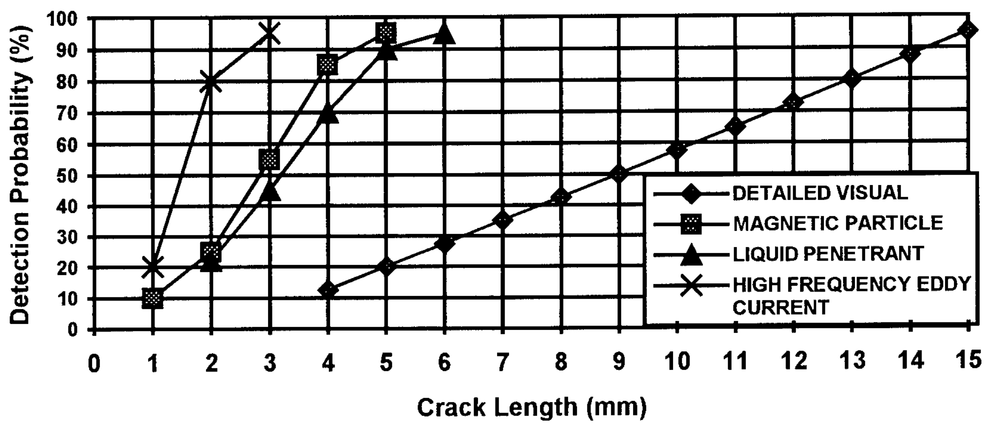


Figure 3. Probability of Detection for Through the Thickness Defects

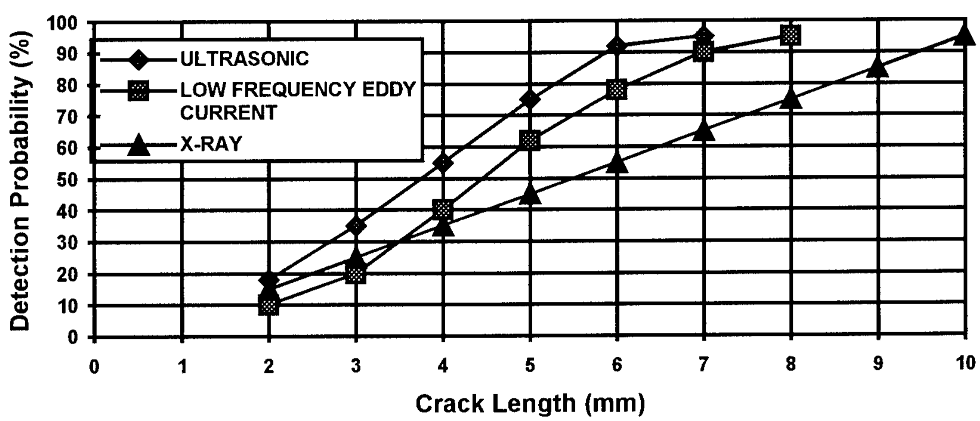


Figure 4. Probability of Detection for Sub-Surface and Internal Defects

4.0 VISUAL INSPECTION

Visual inspection is a sensing mechanism in which eye alone or in conjunction with other aids is used to judge the condition of a component being inspected. Visual inspection is an integral part of airplane maintenance and is considered as a component of NDI. Over 80 % of the inspections on large transport aircraft are visual inspections (Ref. 4). On small aircraft the percentage of visual inspection is even higher. Typical defects found by visual inspection are cracks, corrosion and disbonding. Detection of disbonding by corrosion is generally difficult, however, disbonding may be accompanied by local bulging due to corrosion or entrapped moisture and may be easily detectable.

Visual inspection is perhaps the simplest, most economical and most efficient method of assessing the condition of an aircraft. A large number of defects are generally found by visual inspection and the operators depend highly on the visual inspections to ensure the airworthiness of an aircraft. Hence, visual inspection plays an important role in the safe operation of an aircraft.

4.1 Factors Affecting Visual Inspection

Visual inspection procedures are generally specified by the manufacturer or regulatory authorities in the maintenance or overhaul manuals. A number of factors affect the results of visual inspection. Some of the important factors are:

1) Qualifications and Training of Inspection Personnel- Inspection should be done by qualified personnel or under the supervision of qualified personnel. These personnel should have knowledge of the structural details being inspected, types of defects which are commonly found and the causes of these defects.
2) Inspection Area Access- Proper access to the inspection area is an important factor in the reliability of visual inspection. An easy access to the component to be inspected will assist in the decision making process and ability to interpret results.
3) Lighting- Proper light without glare is essential for a quality visual inspection. Poor lighting can mask the defects and cause fatigue to the inspectors there by affecting their judgment.
4) Pre-cleaning- The part to be inspected should be free from dirt, contamination and any foreign material which will obscure the detection of defects.
5) Working Environment- A proper working environment is necessary for the visual inspectors. Presence of excessive temperature, wind, rain or any other adverse condition can influence the interpretation capability of operators and increase the potential for errors.

4.2 Levels of Visual Inspection

Visual inspection is divided in four categories (Ref. 4), namely: 1) Walkaround Inspection, 2) General Visual Inspection, 3) Detailed Inspection, and 4) Special Detailed Inspection.

Walkaround Inspection-The purpose of a walk around inspection is to serve as a quick check to detect any obvious discrepancies which would affect the performance of an aircraft. Most maintenance manuals specify a walkaround inspection on a periodic basis. This inspection may be done by flight or maintenance personnel from the ground. This inspection includes: fuselage, left and right wings, leading edges, control surfaces, propeller or fan blades, exhaust areas, pylons and gear well. The walkaround is done twice to make sure that nothing was missed the first time. The inspector looks for any major dents in the skin, missing fasteners, corrosion, leaks etc.

General Inspection-A general inspection of an exterior is carried out with open hatches and openings of interior to detect obvious damage. A general inspection is carried out when a problem is suspected or routinely when panels are open for normal inspection. The tools required for this inspection include: flashlight, mirror, droplight, rolling stool, ladder, stand and tools for removing panels.

Detailed Inspection- A detailed inspection is required when a specific problem is suspected or general inspection has identified some problems. This inspection is an intensive examination of a specific area, system, or assembly to detect any damage, failure or discrepancy. Surface preparation and special access may be required for this type of inspection along with special aids in addition to the tools required for general inspection.

Special Detailed Inspection- A special detailed inspection is a thorough examination of a specific component , installation or assembly to detect damage, failure or any discrepancy. Disassembly of sub-components and cleaning may be required for this type of inspection. Tools required for this type of inspection may include flashlight, mirror, borescope, image enhancement and recording devices, rolling stools etc.

4.3 Visual Inspection Equipment

Various aids are used for visual inspection. One of the most important aid in visual inspection is the proper lighting and illumination. Reference 4 describes the ideal lighting and illumination required for proper visual inspection. The reference describes various portable lighting aids. The other inspection equipment required includes: mirrors, magnifiers and equipment to obtain images from inaccessible places being inspected.

Inspection Mirrors- These are used to look at the areas which are not in the normal line of sight. A number of different mirrors are available to inspect hidden areas (Ref. 4).

Magnifying Devices-These are used in the visual inspection to expand the area being inspected for detecting damage and other anomalies. These devices include: simple magnifying glass, microscope and illuminated magnifiers.

Photographic and Video Systems- A photographic image of the area being inspected enhances the decision making capability of an inspector to interpret what he sees. Photographic and video systems are available which can be attached to borescope, fiberscopes or any other visual equipment for documentation and interpretation of visual inspection images. The photographic images can be stored as permanent records for later viewing. A number of systems are available in the market.

Borescopes- A borescope is a tubular precision optical instrument with built-in illumination to allow remote visual inspection of internal surfaces. Borescope tubes may be rigid or flexible and are available in a wide variety of lengths and diameters. These are available in a number of designs and manufacturers can supply custom made borescopes to serve customer needs. The selection of a borescope depends on a particular application and is governed by factors such as- resolution, illumination, magnification, field of view, working length, direction of view, etc.

Borescopes are used in aircraft structures and engine maintenance programs to inspect the areas which are difficult to reach and there by reduce/eliminate costly teardown inspections. These can be used to inspect the interiors of pipes, hydraulic cylinders, turbine blades and valves. They are also used to locate foreign object damage and verify the proper placement and fit of seals, bonds and gaskets.

4.4 Visual Inspection of Composite Structures

The use of composite materials is increasing in aircraft structures due to the improved structural efficiency of these materials and is expected to increase further in future aircraft. Maintaining the structural integrity of these structures is of prime concern to the operators. The in-service damage in composite structures is quite different from conventional metallic structures. In metallic structures detection of cracks and corrosion is of prime concern to the operators whereas in composite structures this kind of damage does not occur.. The most common damage occurring in composites is impact damage which may result in internal matrix cracking, fiber breakage and delamination between plies without any appearance of external damage known as non-visible impact damage. Fortunately, all composite structures are designed for non-visible impact damage.

Any serious in-service damage which may affect the integrity of a structure has to penetrate, chip away or abrade the paint finish of the composite structure. Any damage caused by hail storm, lightning or paint strippers will be easily visible on the surface and can be detected. Once the damage has been detected, the affected area needs to be inspected by other NDI methods for assessing the effect of the damage on structural integrity.

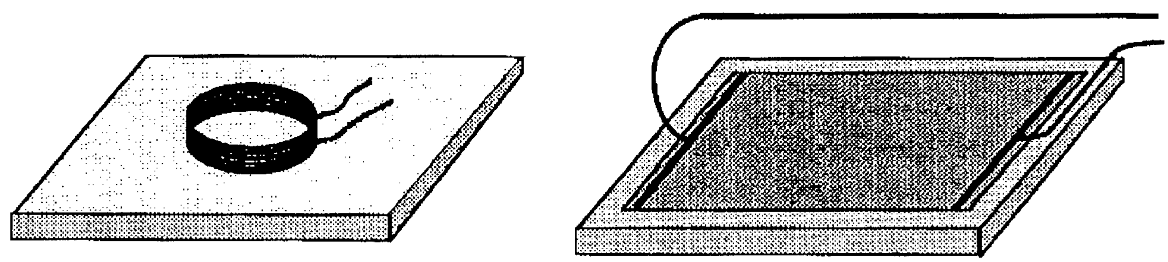
5.0 NON-DESTRUCTIVE INSPECTION METHODS

As mentioned earlier a number of NDI inspection methods are available and the use of a specific method depends on the type of structure being inspected, available access and the desired degree of accuracy in the inspection. Significant advancements have take place in NDI methods recently. The methods and recent advancements are discussed in the following paragraphs.

5.1 Eddy Current

Eddy current is generally used to detect cracks and corrosion near the surface of metallic structures or in thin structures. Eddy current is also used for verifying and separating alloys by differences in their electrical conductivity. This technique has been gradually replacing x-ray. Hand-scanned eddy current probe coils can detect small cracks at fastener holes, however, the method is time consuming and tedious. As most conventional eddy current instruments display variations in the complex impedance, corrected for lift-off as seen by the probe coil, the flaw indications may be sometimes ambiguous. This generally requires trained and experienced operators to interpret the results. Also, the lift-off variations produced by surface roughness or paint thickness can result in false calls. The paint removal may be required prior to inspection with conventional eddy current equipment. Recent trends in eddy current technology have been towards the computerization, automation, improving capabilities to detect small flaws and flaws in multi-layer structures. Two NDI techniques which show significant promise in detection of corrosion and subsurface cracks without disassembly are Magneto-Optic/Eddy Current Imager (MOI) (Ref. 5-8) and Low Frequency Eddy Current Array (LFECA) (Ref. 9-12).

Magneto-Optic/Eddy Current Imager (MOI)- The MOI technique makes it possible to do faster, simpler and more reliable detection of cracks and corrosion in structures. This real-time imaging technology is based on a combination of magneto-optic sensing and eddy current induction. The basic difference between conventional eddy current and MOI technique (Ref. 6) is that in the conventional techniques current flowing in coils is used to induce magnetic field in the test piece whereas MOI produces magnetic field with current in a thin planar foil placed parallel and near the surface of the test piece as shown in Figure 5. A key requirement for MOI is to induce uniform currents in the structure being inspected. As the induced currents are not circular but planar, these are referred to as sheet currents (Ref. 6). In good electrical conducting materials such as aluminum, the currents flowing are small compared to those flowing in coil and can be made to flow uniformly.



Conventional Methods- Coils

MOI- Linear Sheet Current Induction

Figure 5. Conventional and MOI Techniques to Induce Magnetic Field

The images of holes, cracks or other defects are formed as the presence of these discontinuities in a material diverts the otherwise uniform flow of current near the surface of a structure as shown in Figure 6 (Ref. 6). At eddy current frequencies of 25.6-102.4 kHz most through-the thickness fatigue cracks in aluminum are easily detected and imaged, whereas at lower frequencies (e.g. 6.4 kHz) hidden multi-layer cracks, corrosion and substructure (Ref. 5) can be imaged. Figure 7 shows POD of sliding probe and MOI, indicating superior performance of MOI. Figure 8 shows typical cracks detected by MOI and Figure 9 shows corrosion detected by MOI.

The key advantages of MOI are (Ref. 5): 1) Speed of operation 5 to 10 times faster than conventional eddy current, 2) Easy to interpret image formation, 3) No false calls, 4) Elimination of paint or decal for inspection, 5) Easy documentation of results on video or film, and 6) No operator fatigue.

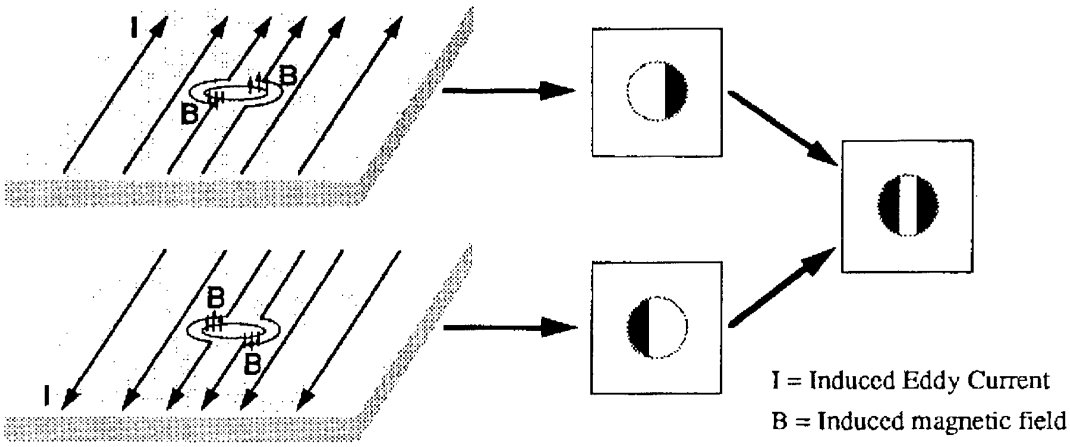
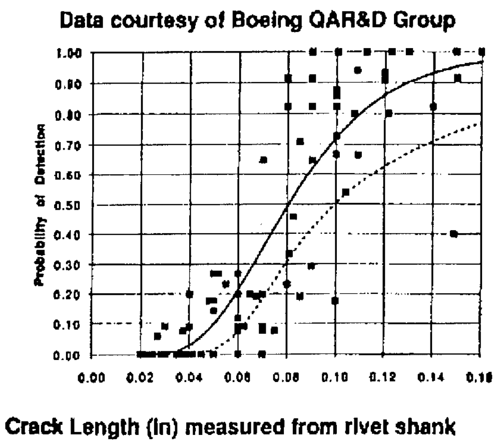
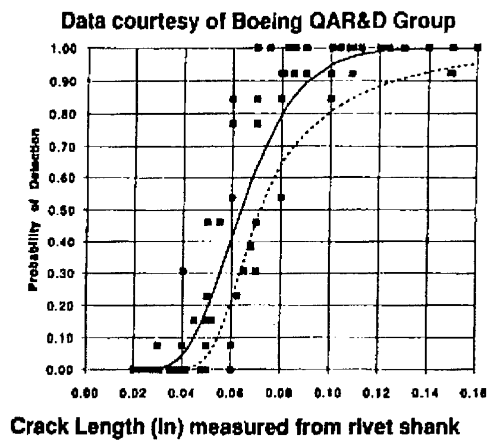


Figure 6. Formation of Images with Magneto-Optic/Eddy Current Imaging

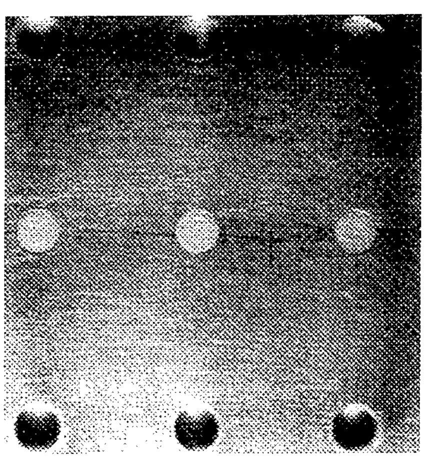


SLIDING PROBE

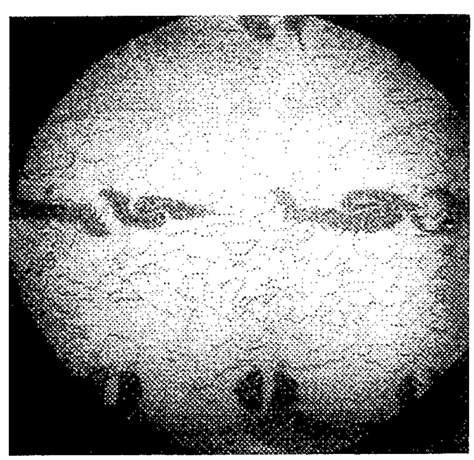


MOI

Figure 7. Detection POD for Sliding Probe and MOI

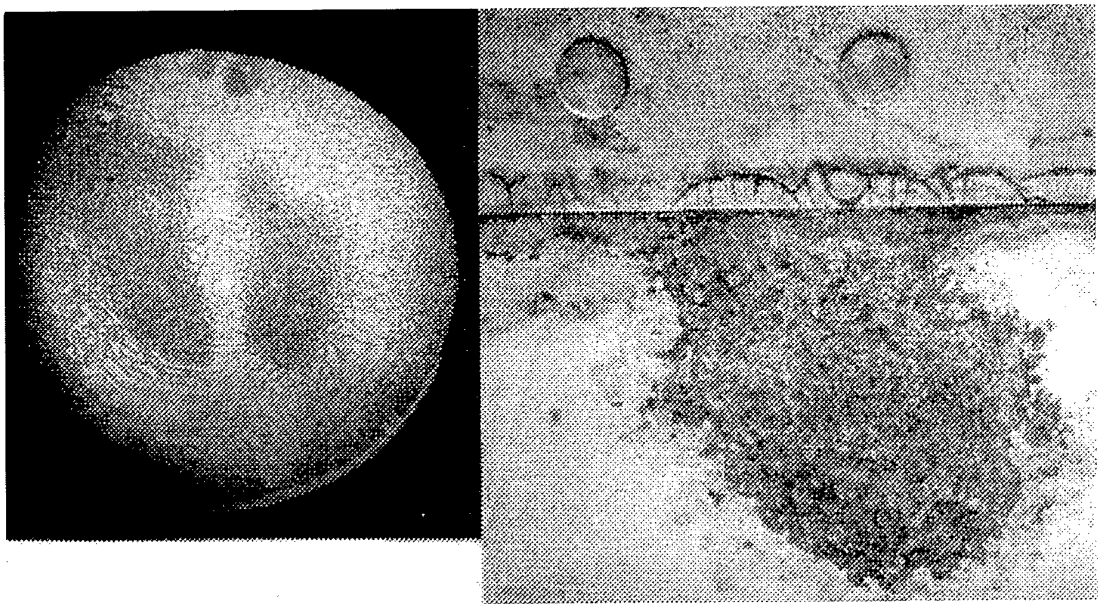


Specimen with Cracks at Fastener Holes



MOI Image

Figure 8. MOI Image of Cracks at Fastener Holes



MOI Image **Specimen with Corrosion**

Figure 9. MOI Image of Corrosion

Low Frequency Eddy Current Array (LFECA)- The LFECA system, developed by the Northrop Grumman corporation, is a portable eddy current inspection equipment to detect subsurface cracks under installed fasteners in multi-layer aircraft structures (Ref. 9-12). The inspections can be performed in near real time without the removal of fasteners. The LFECA system can detect cracks, determine crack length and also give crack depth and orientation. The system consists of a LFECA probe for inspection, shown in Figure 10, three printed circuit boards, a cable and software all assembled in a portable personal computer. The LFECA probe consists of a cylindrical core made from ferrite material with a drive coil located on the center post of this core to generate an eddy current distribution which encircles the fastener being inspected. An array of 16 sense elements, spaced evenly around the outer rim of the core, measures the spatial distribution of these eddy currents. The presence of a crack causes a disruption in the eddy current distribution and is measured by the sense element array. The outer drive coil is used to measure the response due to the adjacent structural features independent of the features at the structural hole. A typical response obtained from the LFECA system is shown in Figure 11 (Ref. 9) for various crack sizes along with the probability of detection. The horizontal tick marks in the figures indicate the 16 angular positions around the fastener hole such that going from left to right will indicate going around the fastener hole once. The horizontal location in the response indicates the orientation of the crack and the magnitude of the peak indicates the crack length.

The probability of detection of cracks with the LFECA system was obtained at Federal Aviation Administration (FAA) NDI validation center at Sandia National Laboratories in Albuquerque, New Mexico, USA (Ref. 9-11). The POD process consists of a blind test of eddy current equipment to inspect a lap joint typical of a commercial airline fuselage shown in Figure 12. The process involves inspection of 43 specimens with each specimen containing 20 fastener holes.

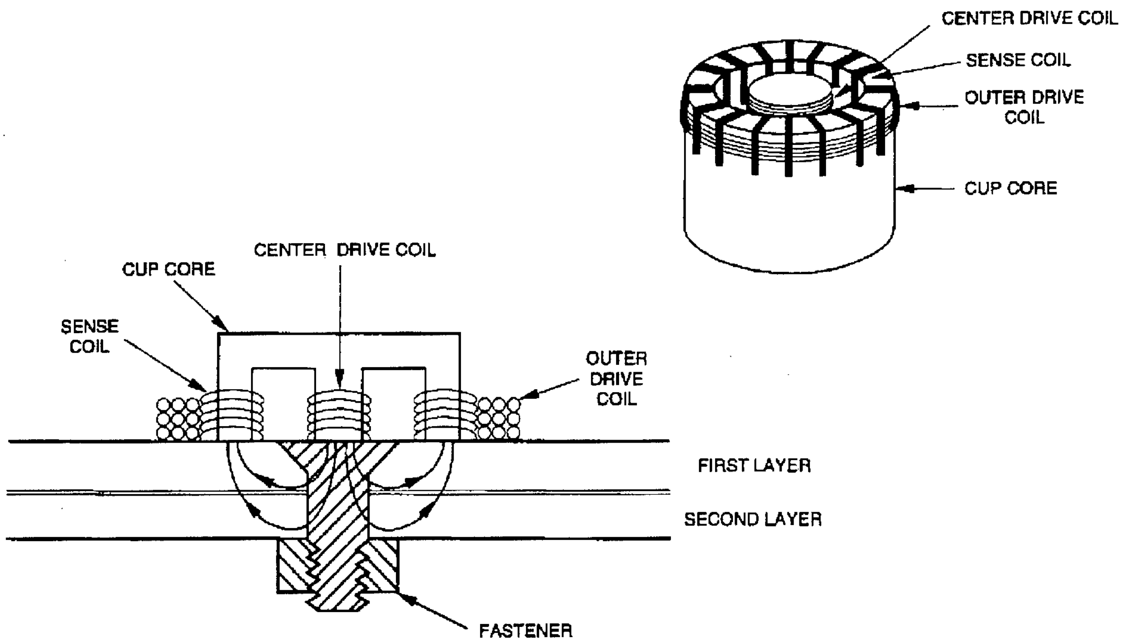


Figure 10. Low Frequency Eddy Current Array Probe

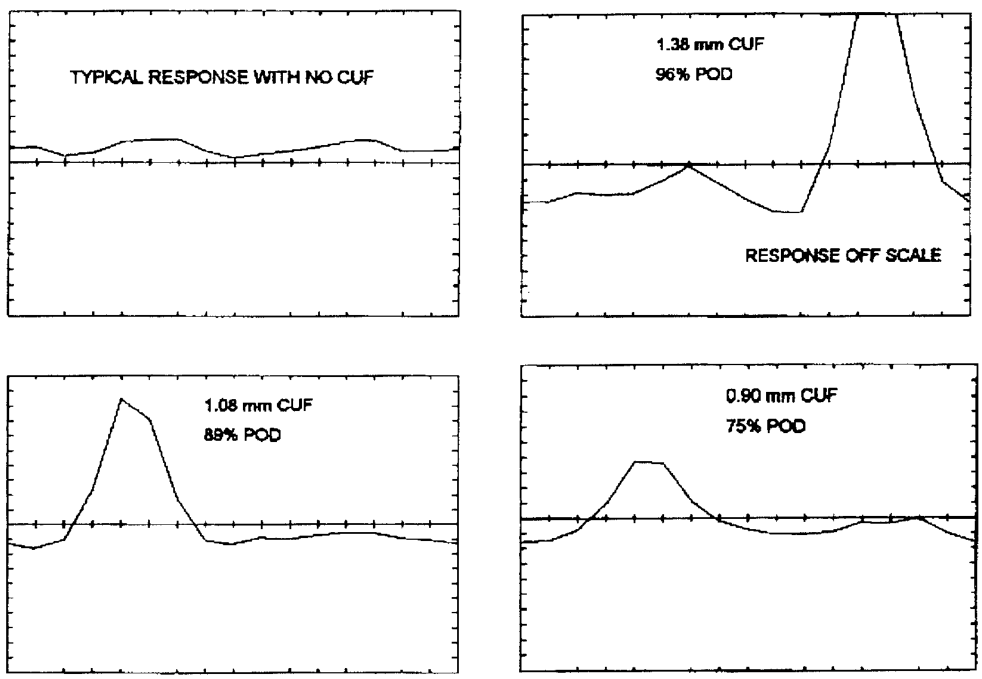


Figure 11. LFECA Response for Cracks of Various Lengths Under Fasteners

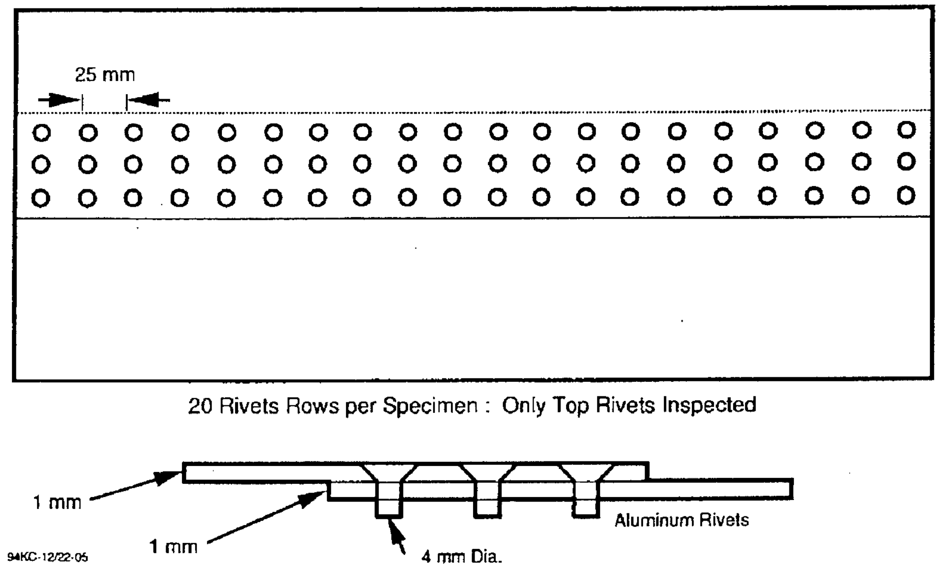


Figure 12. Boeing 737 Lap Splice Specimen Configuration

The specimen were constructed using 1 mm thick 2024-T3 aluminum sheets which were fastened together with three rows of 4 mm diameter aluminum flush head rivets. Fatigue cracks were grown in the first layer of selected holes prior to riveting the panels. A range of crack sizes from 0.3 to 25 mm (a hole to hole crack) were grown within +/- 22 degree orientation (0 degrees being the direction from hole to hole). Holes with cracks on one and both sides were present. Specimens contained either none, a low, a medium or a high number of cracks. A total of 860 holes were inspected with 708 being unflawed holes. The validation exercise contained only the first layer cracks under installed fasteners. Figure 13 shows the POD for the LFECA system and conventional eddy current techniques. It is seen that POD obtained with the LFECA system far exceeds that obtained with the conventional system.

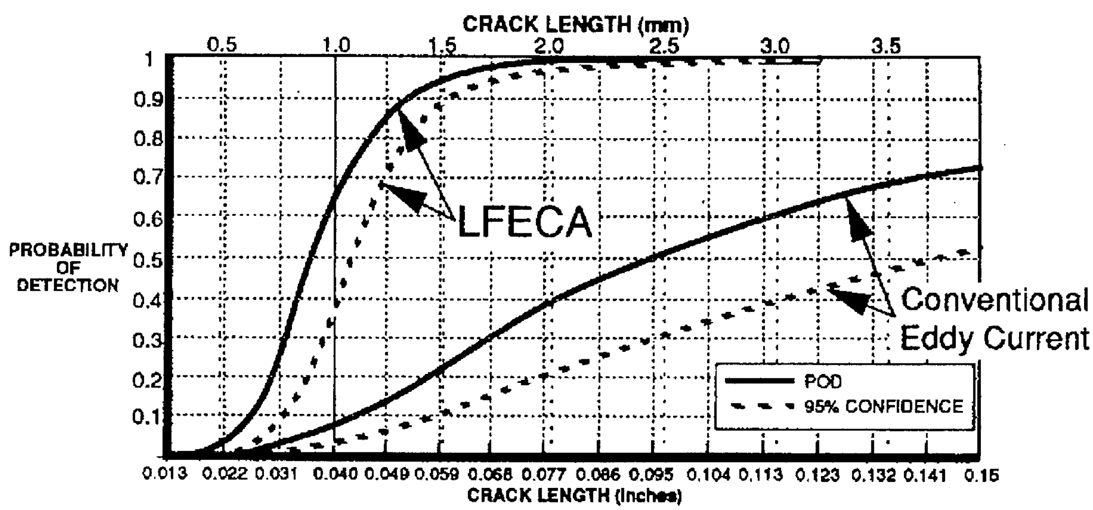


Figure 13. Probability of Detection with Low Frequency Eddy Current Array and Conventional Eddy Current NDI System

5.2 Ultrasonic Methods

Ultrasonic inspection techniques are widely used for quick and relatively inexpensive evaluation of flaws in composite structures. Portable inspection devices are used for on-site inspection of areas with suspected damage. Two methods, namely pulse-echo and through-transmission, are used. In the pulse-echo method, the ultrasound is transmitted by a transducer and the reflected signal is received by the same transducer after the signal has been reflected from the back surface of the composite part being inspected. The attenuation of the reflected pulse is influenced by the presence of the internal defects, and the time delay of the reflected pulse is related to the depth location of the defect. This method is generally used in contact mode of testing and only one side access is required. Inspection of honeycomb structures will require access from both sides for inspection of both face sheets. Ultrasonic inspection using through transmission method is generally conducted with water as a couplant by two methods- 1) Immersion, and 2) Squirting. In the immersion method the part and transducer are immersed in water whereas the squirting method employs a dynamic water column that is squirted and the transducer and the part are suspended. In both methods water acts as the medium that transmits the ultrasound into and out of the part. The images of the defects may be recorded as B-scan, C-scan or 3-D scan. Scans for typical impact damage in a composite part are shown in Figure 14.

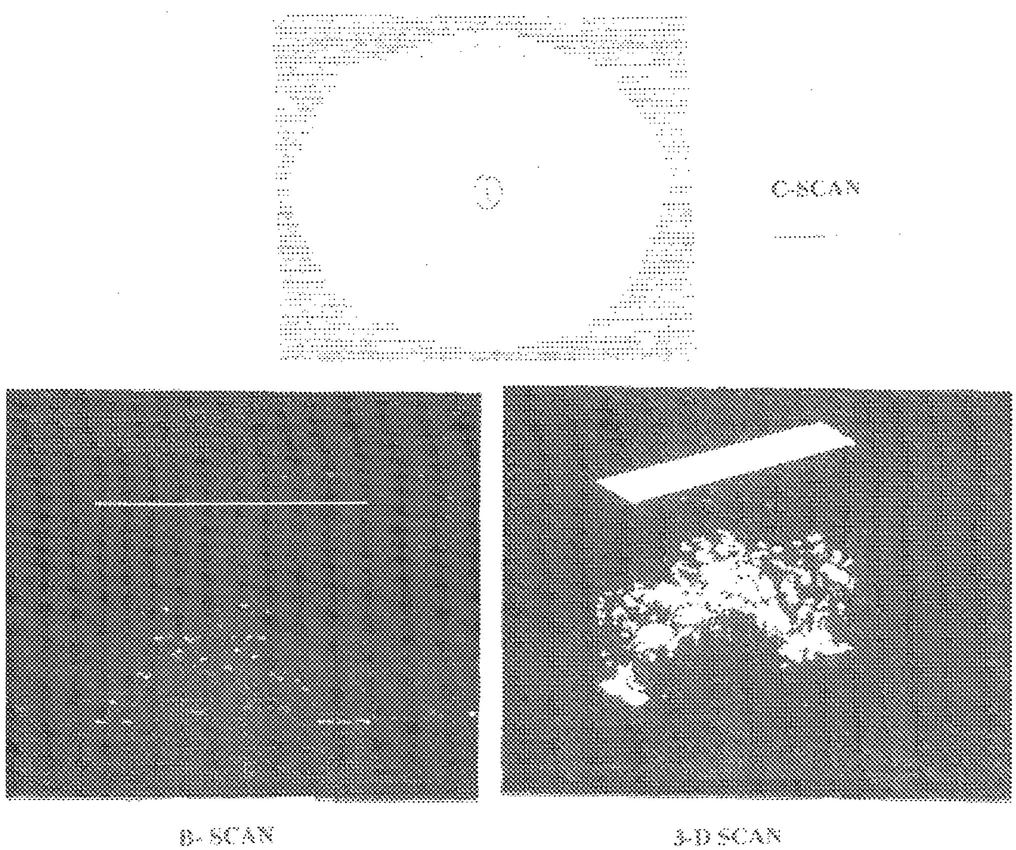


Figure 14. B, C and 3-D Scans of Typical Impact Damage in Composite Laminate

An ultrasonic technique to detect corrosion in a wing box has been developed in Reference 13. The technique has been successfully used to detect corrosion in DC-9 wing box substructure. The current method of inspection is to enter the wet wing box and inspect for corrosion. The technique of Ref. 13 eliminates the entry in the wing box for the inspection and will result in significant savings in the inspection costs.

5.3 Radiographic Methods

The present trend seems to be getting away from using radiographic methods due to safety, cost and maintenance logistics. However, these methods are still being used to detect internal cracks and corrosion in aging aircraft structures. An advanced system known as COMSCAN , developed by Phillips, allows to form images of underlying structure and requires access to one side of the part only. It is currently being used to find corrosion in bulkheads under thin skins, and sonar dome inspections. The system is limited to finding defects near the surface and has the same detection capability as conventional x-ray.

Digiray makes a system which has better resolution and better image quality than the conventional systems. The system is basically the reverse of a conventional digital x-ray imaging system as shown in Figure 15. The x-ray source is formed by a large scanned screen like a TV screen and the detector is a single point sensor as shown in the figure.

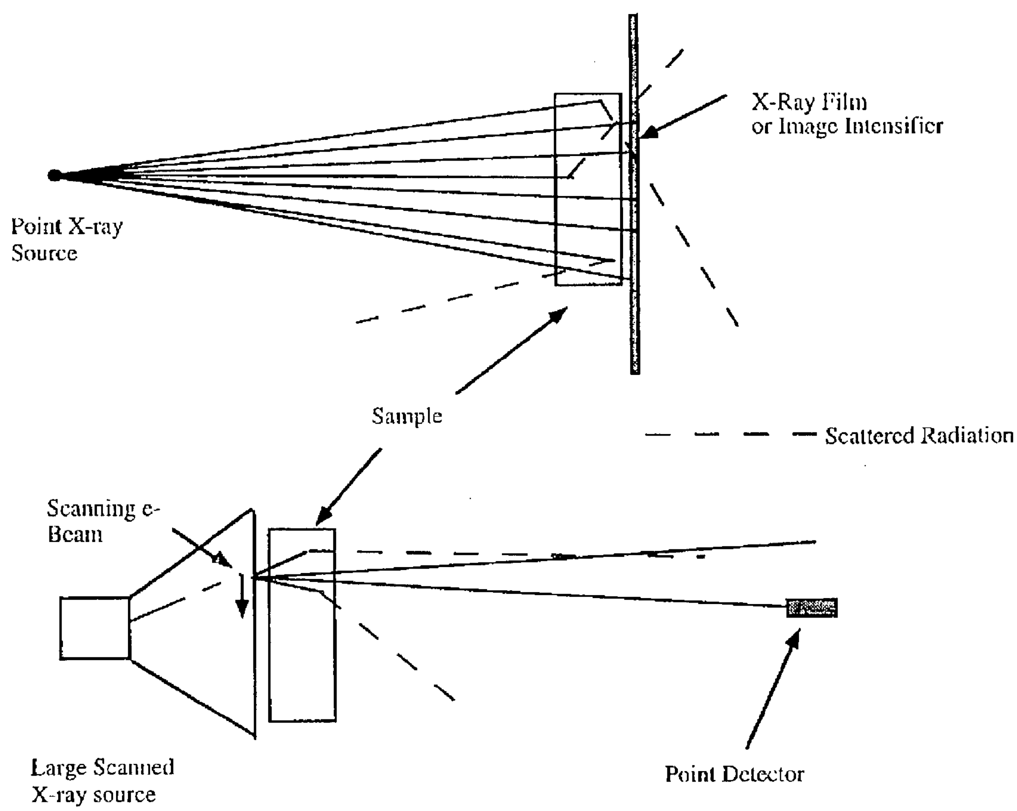


Figure 15. Conventional and Reverse Geometry X-Ray Radiography

5.4 Acoustic Emission

The acoustic emission (AE) technique is used to identify the flaw characteristics by change in acoustic emission signal. Acoustic emission are transient waves that are generated by the rapid release of energy within a material when it undergoes deformation or fracture. This technique has been used to detect damage in composite materials and cracks in metallic structures. Various types of damages in composites such as matrix cracks, fiber/matrix debonding, fiber fracture and delaminations produce acoustic emissions that vary in magnitude, duration and frequency. Various damages in composite materials can be identified by the acoustic emission characteristics. Cracks in aircraft wing were located during ground test with AE technique in Ref. 14 using AE sensors 20 inch (51 mm) apart. However, the source location of flaws could not be precisely predicted.

5.5 Optical Methods

Significant advancements have taken place in optical methods to detect damage in aircraft structures. Some of the techniques being- shearography, DIAS system and thermography.

Shearography- This is a field inspection technique which images internal defects as concentration of surface strain due to an applied stress. A reference image is stored electronically using the shearography video laser interferometer, then a uniform stress is applied in the form of vibration, pressure or thermal, and the subsequent images of the test part are compared with the reference image which will indicate flaws on video monitor (Ref. 15-16). This is a cost effective method for inspection of honeycomb and composite structures. Most of the other NDI techniques do point by point inspections whereas shearography provides a full field video image of flaws in real time. Defects such as debonds, delaminations and impact damage can be detected with this technique.

DAIS- This is a fast and sensitive enhanced visual inspection system for detecting surface irregularities such as pillowing caused by corrosion (Ref. 17-18). In Ref. 17, DAIS system was used in the laboratory as well as in the field to detect corrosion in fuselage lap splices. The results of this reference showed that corrosion pillowing indicative of thickness loss as low as 2% is detectable.

Thermography- This technique uses differential in the thermal conductivity of a defect free part and a part with defects as a basis for locating defects in a structure. A heat source is used to elevate the temperature of the structure being inspected and surface heating effects are observed through a radiometer. For example bonded areas conduct more heat than unbonded areas, the amount of heat either absorbed or reflected indicates the quality of the bond line.

A new technology known as "Thermal Wave Imaging" uses pulses of heat to examine the subsurface in solid objects (Ref. 19). The pulses propagate in the structure being examined as thermal waves and are reflected from any defects, present in the structure, as surface "echoes". These echoes are detected by the use of infrared video cameras, coupled to appropriate hardware and software. The patterns of the echoes on the surface of the structure are used to image subsurface corrosion and disbonds in aircraft structures.

6.0 CONCLUDING REMARKS

Significant advancement has taken place in NDI technology in the recent past. Some of the advancements are discussed in this paper. The use of a particular NDI method is highly dependent on the type of structure being inspected, structural material, desired accuracy, the size of the flaw to be inspected, type of damage, time available and the labor skill. NDI and structural engineers have to make proper choices to assure the reliable detection of the damage with desired accuracy.

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PROSPECTS OF STRUCTURAL HEALTH MONITORING SYSTEMS

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1. INTRODUCTION

Recent trends in research activities have been towards smart vehicle technologies. These activities are primarily geared to design future air vehicles smart so as to perform the roles beyond those performed by conventional vehicles including: 1) Improved performance, 2) Reduced structural weight, 3) Reduced pilot load, 4) Increased survivability and reliability, and 6) Reduced maintenance requirements. These technology areas cover a broad base as shown in Figure 1. From the structural engineer's point of view the key areas of interest are: 1) Avionics/structures integration to reduce structural weight, 2) Smart structures to improve performance, reduce maintenance cost and improve safety of flight, 3) Smart skins to reduce structural weight and improve antenna performance, 4) Infra-red (IR) signature reduction to improve survivability, and 5) Thermal management to improve performance. The majority of the research in these areas is applicable to future aircraft, however, smart structures technology has applications to in-service aging aircraft to assure the safety of flight and reduce maintenance cost.

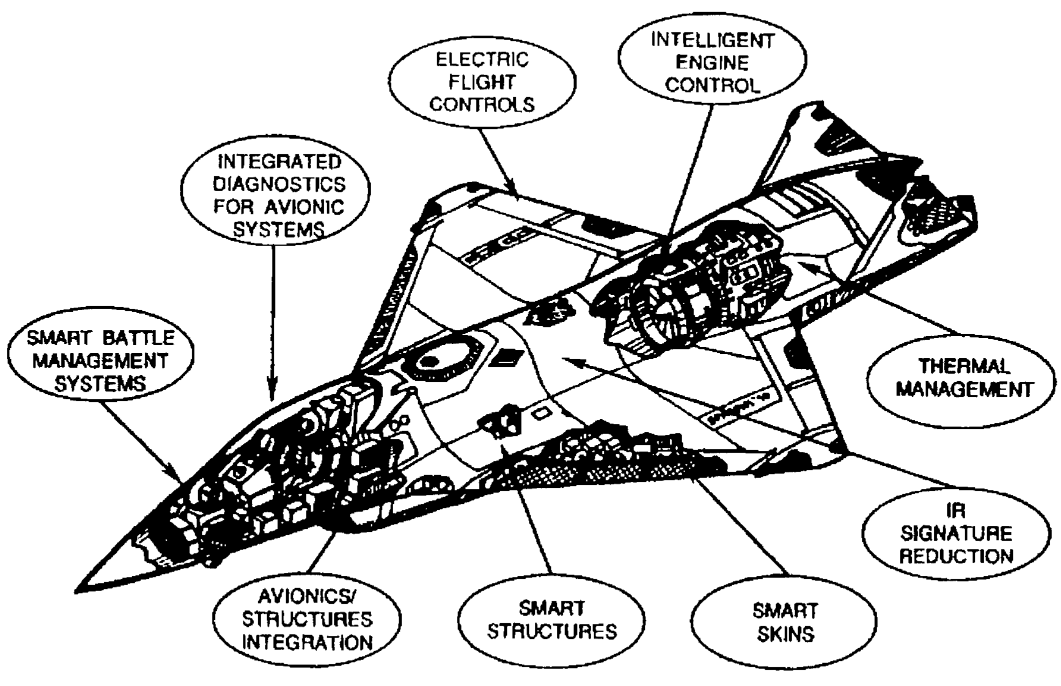

Figure 1. Smart Vehicle Technologies

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The term "smart structures" involves a broad range of concepts, technologies and applications (Ref. 1-7). The two concepts of interest in aircraft are :1) Active or adaptive structures, and 2) Automated structural health monitoring systems. The purpose of automated structural health monitoring (SHM) system is to evaluate structural integrity in real time. Recent advancements in sensors and computers have made the development of automated SHM system feasible.

2. STRUCTURAL HEALTH MONITORING PROCEDURE

A structural health monitoring system should consider both in-service and battle damage. The SHM procedure (Ref. 1) is outlined in Figure 2. The real time monitoring functions will include battle damage as well as any significant in-service damage. Such damage will cause redistribution of loads in the structure. The long term monitoring will include: impact damage, cracks and corrosion.

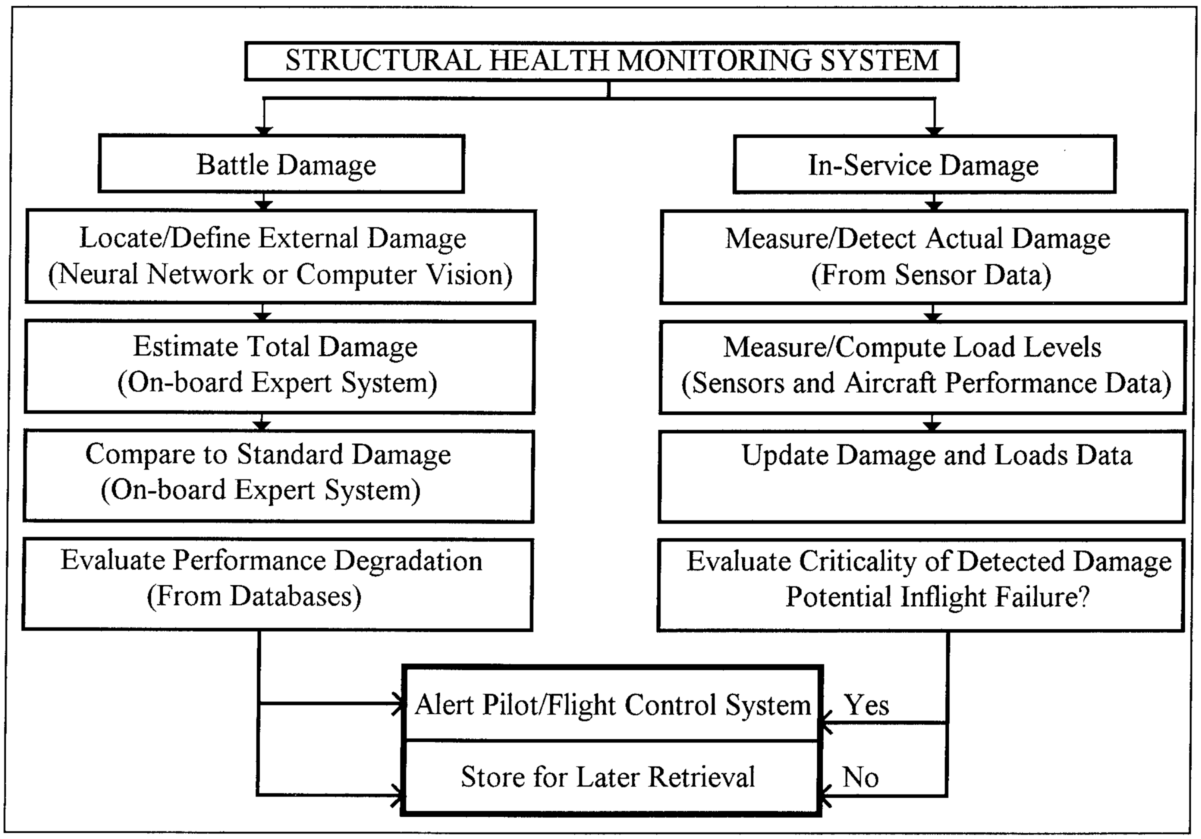

Figure 2. Structural Health Monitoring Procedure

Significant structural damage due to foreign objects, particularly in a battle environment, will cause load redistribution in a structure and the information can be used to determine the extent of the damage. A neural network can be used (Ref. 8) to recognize the type and the extent of damage by measuring the strain values at a certain number of discrete locations due to the load redistribution. A large area damage can be detected by optical or acoustical methods. After the large area damage has been detected, the structural degradation and performance can be evaluated by using an on-board expert system which has built in capabilities to evaluate the effect of standard damages on structural performance. The expert system will compare the detected damage with the standard damage and evaluate the performance, thus, avoiding the need for real-time global structural analysis.

The in-service damage such as cracks, corrosion, impact damage, delaminations, etc., are detected by using appropriate sensors. The sensors should have the capability to detect the extent and the location of the damage. After the flaws are detected, their criticality is governed by ASIP/NASIP requirements to make decisions on repair, inspections etc. The criticality analysis of these flaws will require the knowledge of local loads, material properties and structural details. This is not an easy task and a simplified approach needs to be developed. One simple approach is to store in the expert system the critical flaw sizes for all known critical locations from stress reports, full scale fatigue tests and in-service experience. Once the damage is located at a location, the expert system will compare the damage with the built in library and make suggestion of the appropriate action to be taken. If a flaw is located at an unexpected location, the expert system will compare the criticality with a nearby location and determine criticality and make suggestions for further analysis at the location.

3. STRUCTURAL HEALTH MONITORING SYSTEM (SHMS) ARCHITECTURE

Various components of smart health monitoring system include: 1) Sensors, 2) Local and central processors, and 3) Software capable of making intelligent decisions. A typical smart health monitoring system architecture is shown in Figure 3 (Ref. 1,5).

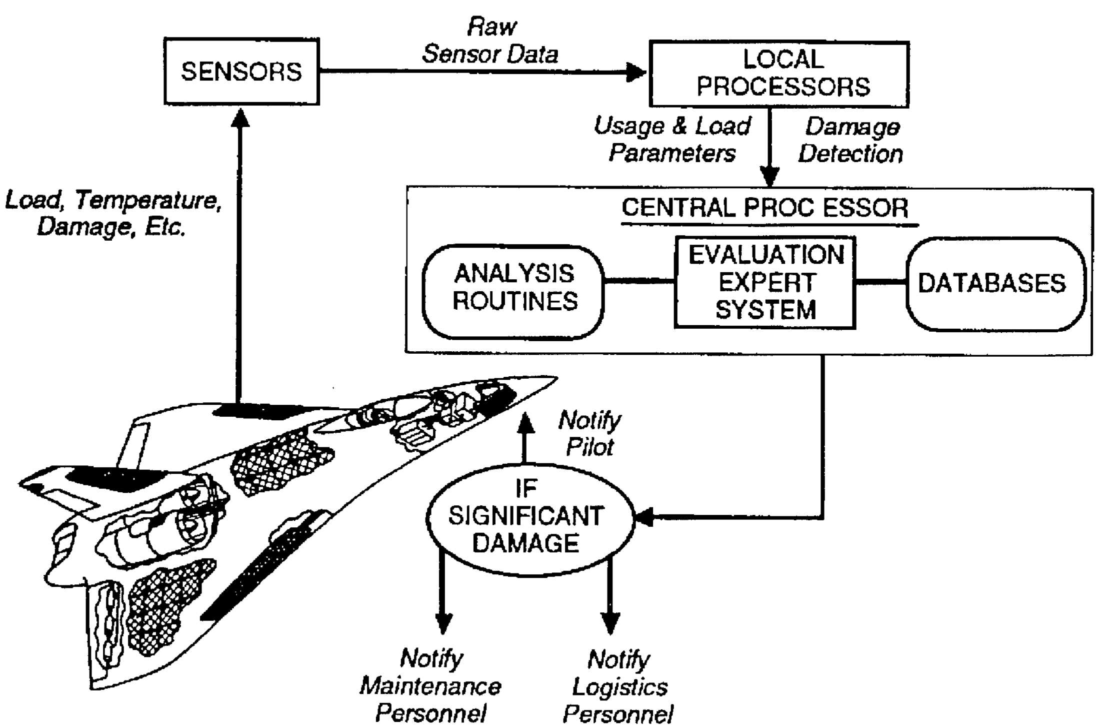

Figure 3. Structural Health Monitoring System Architecture

Sensors- One of the key components of structural health monitoring are the sensing devices to detect damage and loads information. The success of the SHMS is highly dependent on the use of proper sensors, their capability, and the proper location on the aircraft structure. A list of commonly used sensors and their applications (Ref. 9) are shown in Figure 4.

SENSOR	APPLICATION
Acoustic Emission	Damage Detection, Tracking
Strain gage	Strain Measurement
Fiber-optic	Strain, Temperature, Pressure
Crack Gage	Crack Growth
Fatigue Fuse	Fatigue Life
Accelerometer	Aircraft C.G Loading
Thermocouple	Temperature, Heat Transfer
Pressure Transducer	Pressure
Displacement Transducer	Structural Deformation, Control Surface Deflection
Electro-chemical	Corrosion, Corrosive Environment

Figure 4. Structural Health Monitoring Sensors and Their Applications

A health monitoring system will use acoustic emission sensors to detect structural damage, strain gages and fiber optic gages to monitor strains and temperature, and electro-chemical sensors to detect corrosion. The acoustic emission sensors have shown promise for detecting damage from remote location on the aircraft i.e. without being at the location of damage. The aircraft usage will be monitored by commonly used sensors such as accelerometers, pitot tubes and deflection transducers.

Processors- The data, obtained from sensors (acoustic emission, strain gages etc.), will be controlled by local processing units. The local processor acquires the data, digitizes the data and stores it for subsequent retrieval by the central processor. The central processor has analysis routines, evaluation expert system and databases. It performs the health assessment and recommends the necessary action to be taken.

Software- The software for SHMS includes data collection, databases, analysis algorithms and expert system. The software should be capable of collecting sensor data, analyze the data, and make decisions regarding inspections, maintenance and repairs (Ref. 3).

4. STRUCTURAL HEALTH MONITORING SYSTEM SENSOR EVALUATION

The acoustic emission (AE) technique has shown a good promise for detecting damage in aircraft structures. In Ref. 10 the AE technique was used to remotely detect cracks on an F/A-18E/F titanium bulkhead structural test. The test article, measuring 100 inches (2540 mm) by 60 inches (1524 mm), was the fuselage station 491 titanium bulkhead (Figure 5). The structural fatigue test of the test article consisted of 12,000 hours (2 lifetimes) of spectrum fatigue followed by 6000 cycles of constant amplitude fatigue testing.

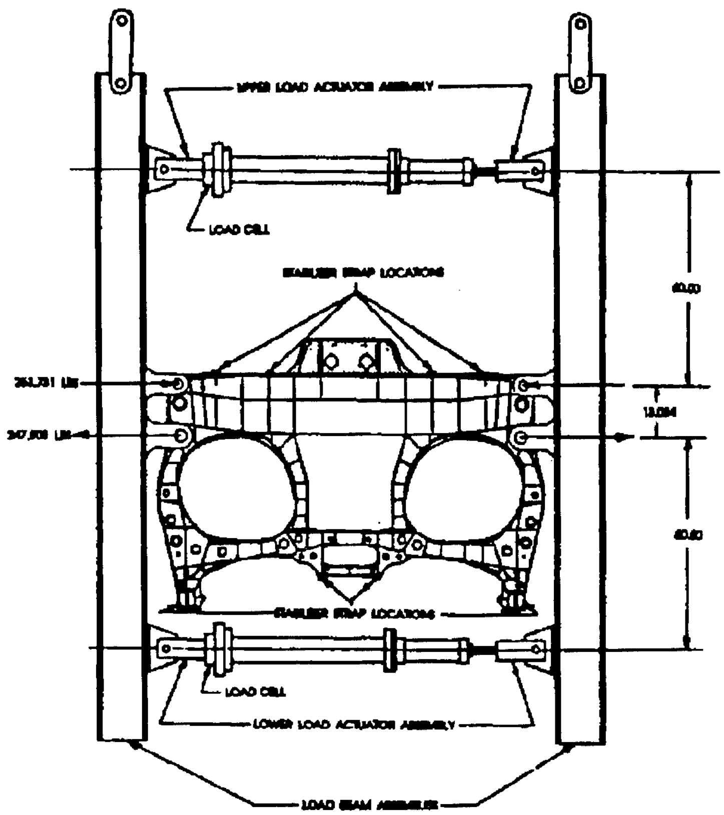

Figure. 5 F/A-18 E/F Fatigue Test Article

AE sensors were used to instrument the bulkhead and were located where the cracks were most likely to occur. No cracks were detected by the AE sensors. Tear down inspection of the test article after the first and second life time of spectrum fatigue loading showed no fatigue cracks. However, during the subsequent constant amplitude test fatigue cracks appeared at access cut out after 1000 cycles. These cracks could not be initially detected as they initiated on the side opposite to where AE sensors were placed. Relocating the sensors (Ref. 10) on the side of the bulkhead where cracks initiated (Figure 6) enabled credible observation of the cracks and the noise signal of the surrounding structure.

5. STRUCTURAL HEALTH MONITORING SYSTEMS BENEFITS AND PAYOFFS

A SHMS system will reduce the need for many labor intensive tasks such as the nondestructive and tear down inspections, resulting in life cycle cost savings. The system will detect any unforeseen damage and thus increase the safety of flight.

Preliminary studies of Ref. 9 have shown that significant savings can be achieved by eliminating some of the manpower requirements associated with inspections. For the F-18 aircraft, savings in excess of $35 million were projected based on 1000 aircraft and usage of 33 hours per month.

The estimated savings for T-38 aircraft were $9 million per year based on 720 aircraft and usage of 420 flight hours per year.

One big advantage of SHMS is the avoidance of catastrophic failures due to the unforeseen in-service structural damage or battle damage. The savings due to the prevention of catastrophic failure are difficult to estimate, but the prevention of one such failure will offset the cost of SHMS system for the entire fleet.

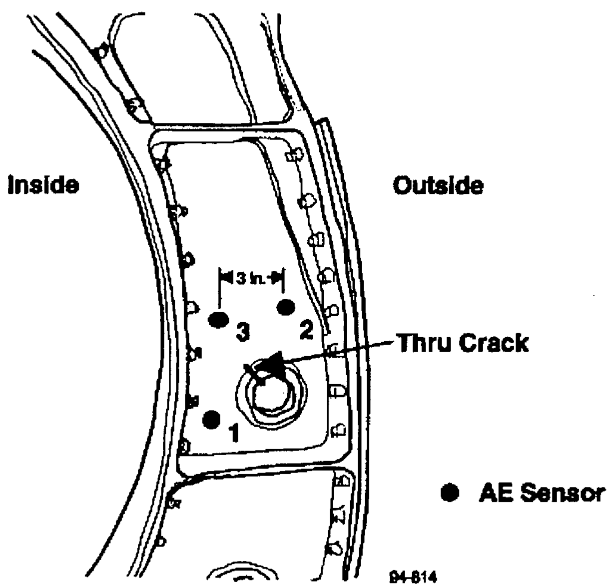

Figure 6. Acoustic Emission Sensor Location

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DURABILITY AND DAMAGE TOLERANCE

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SUMMARY

Durability and damage tolerance are two of the most critical elements of a structural integrity program for aging aircraft. Durability is primarily concerned with economic issues, dealing with maintenance and repair requirements and their associated costs. Damage tolerance is primarily concerned with structural safety, precluding the occurrence of catastrophic failure. Both durability and damage tolerance can affect the operational readiness of aging aircraft. Significant issues involving durability and damage tolerance are widespread fatigue damage (WFD) and structural repairs. This paper considers three topics of widespread fatigue damage. The first topic describes equivalent initial flaw size (EIFS) technologies that characterize the initial fatigue quality of structural details. Such EIFS distributions can be used in durability and damage tolerance analyses. The second WFD topic addresses the contributions of fretting fatigue to the formation of multiple-site damage (MSD) and the effects of such MSD on the remaining structural life and residual strength of aircraft structural joints. The third WFD topic describes a probabilistic analysis that assesses the risk of catastrophic failure. This paper also considers three structural repair topics. The first topic describes an analysis approach, based on the Finite Element Alternating Method, that ensures the structural integrity of adhesively bonded composite repairs of damaged metallic structure. The second repair topic presents the successful application of a composite (born/epoxy) repair of metallic (7075-T6 aluminum) C-141 lower-wing structure. The third repair topic identifies the advantages of using the unidirectionally-reinforced glass fiber/aluminum laminate GLARE®2 as a bonded repair of fatigue-damaged fuselage structures.

1. WIDESPREAD FATIGUE DAMAGE

1.1 Equivalent Initial Flaw Size Technologies

The probabilistic-based equivalent initial flaw size (EIFS) approach [1,2] is useful for aging aircraft applications, including risk assessments, multiple-site damage (MSD), multiple-element damage (MED), widespread fatigue damage (WFD), life extensions, durability and damage tolerance analyses, etc. A broad-brush review of EIFS technologies for aging aircraft is presented. Useful applications are described. Essential elements of EIFS technologies, key issues, and future research needs are discussed. Example results are also presented.

Several useful applications of the EIFS approach and the type of information obtained are summarized in Table 1 [3]. This approach can be used for reliability-centered maintenance analysis and to quantitatively assess maintainability and supportability requirements for metallic aircraft parts, components, or airframe structure. It is also useful for determining the EIFS distribution at critical locations required in a risk assessment analysis [4].

An equivalent initial flaw size distribution (EIFSD) characterizes the initial fatigue quality of a structural detail due to material and manufacturing variations. An EIFS value is determined by growing a crack of size x_1 at service time τ backwards to time zero (i.e., t=O), using well-established fracture mechanics principles (see Fig. 1). The crack size x_1 at service time τ may be obtained from inspection maintenance results or laboratory tests. The EIFS methodology has been developed [5, 6] for establishing the EIFSD for different critical locations in a metallic part or component. The EIFSD can be established for a reference critical location using a limited amount of baseline test data. This methodology accounts for the effects of

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material and manufacturing quality, stress risers (e.g., notches), stress gradients, specimen size, flaw type, etc., on the EIFSD for different types of structural details. Once the EIFSD has been established, the distribution of crack size at any service time t and the distribution of service time to reach any crack size x can be predicted using a crack growth approach based on fracture mechanics principles.

The probability density function for the crack size at any service time t is shown in Fig. 2. The probability of exceeding any crack size x_1 at service time τ is denoted by $p(i,\tau)$. This is an important quantity for all applications of the EIFS approach. The quantity $p(i,\tau)$ is represented by the solid area under the probability density function at service time τ above the crack size x_1.

Considerable progress has been made since the early 1970's in developing the EIFS approach and obtaining a better understanding of the technical problems and limitations which must be addressed to acquire the desired EIFS technologies. To develop the desired EIFS technologies and to gain wide acceptance by the aerospace community, the following key issues must be resolved: (i) the generic nature of EIFS (e.g., dependency of EIFS on stress level, load spectra, flaw type, short crack behavior, stress intensity factor, etc.), (ii) the analytical tools which account for the effects of various factors (e.g., specimen size, type of flaw, stress gradient, etc.) on the EIFS cumulative distribution for critical structural details, (iii) the required test data for assessing the analytical tools and (iv) a convincing validation of the EIFS technologies for practical applications of interest.

Theoretically, EIFSs should be generic but this needs to be verified using experimental data. Methodology has been developed for computing the EIFSD for different critical locations from an EIFSD for a reference location [5, 6]. This methodology accounts for the effects of material and manufacturing variations on the EIFSD. It has also been demonstrated using S-N data for simple types of specimens (i.e., smooth unnotched, open-hole and double-edge notched) with polished surfaces to emphasize the effects of initial material quality on the EIFSD. Further research is required to demonstrate the methodology using experimental data for structural details reflecting both material and manufacturing effects on the EIFSD. Due to the complex nature and many facets of the problem, the research will focus first on developing mechanistic-understandings for relatively simple types of structural details (e.g., open hole, semi-circular edge notch, etc.). The overall objective is to develop and verify EIFS technologies applicable to full-scale aircraft parts and components - not only for general design applications but also for aging aircraft. The EIFS technologies should be established in a building-block fashion with progress measured one step at a time. Rather than attacking the most complex applications of interest (e.g., any type of structural detail in manufactured aircraft parts and components subjected to applicable environments, including the effects of corrosion), the focus should be on the simple cases first, with progression toward the complexities of interest.

1.2 Fretting Fatigue and Multiple-Site Damage (MSD)

Basic research directed toward analyzing widespread fatigue damage (WFD) includes fretting fatigue and multiple-site damage (MSD) studies. The goal of the fretting fatigue work is to predict the onset of small cracks at fastener holes or other structural members with contacting surfaces. Experiments and analyses are directed at determining the effect of various parameters on the local contact stresses. Multiaxial fatigue theory then relates the localized stress state to crack formation life. Once regions of MSD have formed by fretting (or other sources of cracking), the next goal is to determine the consequences of such MSD on the remaining fatigue life and residual strength of a structural joint. A model for the load transferred by mechanical fasteners has been developed and incorporated into a fracture mechanics analysis for the crack growth, coalescence, and final fracture of the joint. Further details of this work are described below.

Fretting is a wear and fatigue phenomenon occurring between contacting surfaces nominally at rest having relative oscillatory motion of small amplitude. The nature of load transfer in mechanical joints leads to fretting which is characterized by micro-slip at the edges of the contact surfaces and the cyclic contact stresses. Fig. 3 illustrates how remote cyclic loading of a lap joint causes both a normal pressure p(x) and shear traction q(x) at the fastener holes. A numerical model has been developed for the load transferred at various fastener locations. This load transfer model includes the effects of fastener/joint material, plate thickness, fastener sizes, and hole patterns. Finite element analyses may then be used to determine the local fretting stresses at various contact points in the structure of interest.

To relate these fretting stresses with crack nucleation life requires a multiaxial fatigue model based on experimental data. As shown in Fig. 3, the fastener/skin contact is approximated in the laboratory by an applied tangential force Q, which must be less than the friction force μP required to cause global sliding. These conditions lead to a region of stick, spanning the middle of contact with regions of micro-slip on either side. This stick/slip region is being examined experimentally with a fretting fatigue rig which applies controlled normal and tangential forces to the fatigue specimen. These laboratory specimens are being used to develop a multiaxial fatigue life model that accounts for the effects of various contact parameters, including the magnitude of the normal and tangential loads transferred though the fastener, the friction coefficient at the interface, and appropriate material properties.

Once the fretting fatigue analysis indicates that crack nucleation (typically a crack which is one mm in length) has occurred, a global MSD analysis determines the remaining life of the structure. At this point in time, it is assumed that the fastener holes are cracked due to fretting (or by other mechanisms), and a fracture mechanics analysis is used to determine the cyclic growth of individual cracks, their link-up with cracks from adjacent holes, and subsequent failure. Fig. 4 shows the fatigue crack growth results for a typical lap-joint test specimen. In this case, two 2024-T3 aluminum sheets were riveted together with three rows of 4.8-mm diameter 2117 aluminum rivets. One row of rivet holes was precracked before assembly, and then the assembled specimen was cycled to failure under constant amplitude loading. Crack extension from each hole is shown in Fig. 4 as a function of elapsed cycles. (Open symbols are experimental measurements, while the solid lines are the numerical predictions.) Note that the numerical model does a good job of predicting the growth, coalescence, and final failure resulting from the individual cracks. Current MSD work deals with incorporating the results of stiffeners in the analysis and on conducting experiments with larger specimens to further verify the analysis techniques.

1.3 Structural Risk Analysis

The United States Air Force implements durability and damage tolerance requirements through the Aircraft Structural Integrity Program (ASIP). Because of ASIP, deterministic durability and damage tolerance evaluations and individual aircraft tracking have been performed for the critical locations on all Air Force aircraft. To provide an additional tool for making decisions regarding the scheduling of inspections, repairs, and replacements, a computer program has been written to perform stochastic risk analyses for aging aircraft fleets. This program is entitled PRobability Of Fracture (PROF).

PROF is a computer program that runs in the Windows environment on a personal computer and was specifically written to interface with the data that are available as a result of ASIP. Fig. 5 is a schematic of the program which illustrates the types of data required to perform a risk analysis and the probability of fracture (POF) output that is calculated as a function of flight hours. Under ASIP, fatigue crack growth predictions (i.e., crack size versus time, a versus t) are available for every known critical location. This implies the availability of: a) the flight-by-flight stress spectrum, from which the distribution of maximum stress per flight can be obtained; b) stress intensity factors as a function of crack size, a versus K/S; and c) fracture toughness data, $K_{cr,}$ from which a distribution of fracture toughness can be inferred. The initial crack size distribution can be obtained from inspection feedback, tear-down inspections, or equivalent initial flaw sizes. Probability of detection as a function of crack size, POD(a), is a characterization of the capability of the nondestructive inspection system used during the safety inspections.

The starting point of a PROF analysis is a representative distribution of initial crack sizes. PROF uses the deterministic a versus t curve to project the percentiles of the initial crack size distribution as a function of flight hours. At defined flight hour increments, the single flight probability of fracture is calculated from the distributions of crack size, maximum stress per flight, and fracture toughness. That is, the single flight fracture probability is the probability that the maximum stress intensity factor (combination of the distributions of maximum stress per flight and crack sizes) during the flight exceeds the critical stress intensity factor.

After maintenance and repair actions have been accomplished, the distribution of crack sizes is changed in accordance with the POD(a) function and the equivalent repair crack size distribution. It is assumed that all detected cracks are repaired and the equivalent repair crack size distribution accounts for the repaired cracks. PROF produces files of both the pre- and post-inspection crack size distributions. The availability of these distributions allows changing the analysis conditions at inspection times set by the analyst.

The a versus t, a versus K/S, and crack size distributions are input to PROF in tabular form. Fracture toughness is modelled by a normal distribution and requires values for the mean and standard deviation. Maximum stress per flight is modelled by the Gumbel extreme value distribution and the parameters of the distribution can be obtained from a fit of either a flight-by-flight stress spectrum or an exceedance curve of all of the stresses in the spectrum. The POD(a) function is modelled by a cumulative lognormal distribution with parameters m and s. Fifty percent of the cracks of size m would be detected. The parameter s determines the flatness of the POD(a) function, with smaller values implying steeper POD(a) functions.

Sensitivity studies have been performed on the application of PROF in representative problems. These studies have indicated that, although the absolute magnitudes of the fracture probabilities are strongly dependent on the input, relative magnitudes tend to remain consistent when factors are varied one at a time. Because of the indefinite nature of some of the input data, particularly the crack size information, absolute magnitudes of the fracture probabilities are suspect. However, it is believed that relative differences resulting from consistent variations in the better-defined input factors are meaningful.

A single run of PROF analyzes the growth of a crack for a single geometry, including crack type and shape. The analysis would apply to the population of structural details which both have this geometry and are subjected to an equivalent stress spectrum. The output of PROF includes fracture probabilities for a single structural detail, for a single aircraft when there are multiple equivalent details, and for the entire fleet. The inspection intervals are set by the analyst, including the possibility of an immediate inspection at time zero.

More complex problems can be analyzed by combining the results of multiple runs. First, intermediate output can be used to initiate new runs for changed conditions. Examples of such analyses would include the introduction of corrosive thinning of the material, the effect of oversizing holes during repairs, and the effects of changing aircraft mission usage. The results from multiple runs for different details can also be combined to model more complex scenarios. Examples of such scenarios include the analyses of multi-element and multi-site damage.

2. REPAIRS

2.1 Analysis Method for Adhesively Bonded Composite Repairs of Metallic Structure

2.1.1 Introduction

Adhesively bonded composite repairs of metallic structure have many advantages over riveted metal repairs. These include: (i) no introduction of new stress concentrations into the repaired structure due to rivet holes; (ii) the composite patches are readily formed into complex shapes; (iii) high stiffness-to-weight and strength-to-weight ratios of the patch; (iv) high corrosion and fatigue resistance of the composite; and (v) inspection via eddy-current is possible for non-conducting fiber systems. To exploit this technology to the fullest, the United States Air Force has initiated several developmental programs. These programs are in the areas of bonded repair design, application and inspection. This section will deal with one of the efforts in the area of design.

In order to properly design an adhesively bonded composite repair of a metallic structure, many factors must be considered. These include:

- global stiffening of the aircraft structure due to the high stiffness of the composite patch;
- the effect of size, shape, thickness and material properties of the composite patch on the crack-tip stress intensity factors;
- the effect of the material properties of the adhesive on the crack-tip stress intensity factors;
- the effect of thermal cycling on the composite repair; and
- the effect of disbonds on the effectiveness of the composite repair.

To address these design factors, software which implements a Finite Element Alternating Method (FEAM) based methodology is being developed [7]. The FEAM is a powerful technique which can be used to efficiently obtain the stress intensity factors associated with a crack in a finite body. The advantages of utilizing the FEAM to compute stress intensity factors are many. However, they are all a result of the fact that only the uncracked structure is modeled with finite elements. As a consequence, the FEAM is extremely efficient from both a computational and manpower point of view when performing parametric studies of crack size and location because the finite element mesh remains the same. Note also that this property makes the FEAM ideal for performing fatigue crack growth calculations.

2.1.2 Software Description

The software development plan consists of six tasks. At present, Task 1 has been completed. The other tasks will be carried out on a priority basis.

2.1.2.1 Task 1

Software for the repair of surface and corner cracks in monolithic aircraft structure has been completed. The ability to analyze the effect of size, shape, thickness, tapering and material properties of the composite patch on the crack-tip stress intensity factors and adhesive shear stresses is provided. This software package consists of two programs, PATGEN_3D and COMPAT_3D. PATGEN_3D serves as a special purpose pre-processor for COMPAT_3D, which performs the actual fracture mechanics computations.

To minimize the learning curve associated with using COMPAT_3D, input data in the form of PATRAN neutral files and NASTRAN bulk data files can be used. Thus, the user need not learn a new finite element pre-processor. The element types presently supported by COMPAT_3D are:

- twenty-node bricks for modeling the monolithic metallic structure;
- eight-node shear elements for modeling the adhesive layer; and
- eight-node plate elements for modeling the composite patch.

COMPAT_3D is written in standard FORTRAN 77. Hence, it is easily ported to various computer platforms. The list of supported systems includes:

- IBM compatible PC's,
- Hewlett-Packard UNIX Workstations,
- IBM UNIX Workstations,

- Sun UNIX Workstations,
- Silicon Graphics UNIX Workstations, and
- Digital UNIX Workstations.

The steps involved in using COMPAT_3D are as follows [8].

1. Build a finite element model using your favorite pre-processor (which can write a PATRAN neutral file or NASTRAN bulk data file).
2. Use PATGEN_3D to translate the finite element model developed in step 1 and create the COMPAT_3D input file.
3. Run COMPAT_3D.
4. Utilize the COMPAT_3D output (stress intensity factors, crack growth lives, etc.) in a damage tolerance analysis.

2.1.2.2 Task 2

The capability to analyze the repair of through-the-thickness cracks (including WFD) in stiffened aircraft structure will be added to the software in Task 2. This addition to the capability provided in Task 1 will allow for the seamless analysis of a crack as it transitions from a part-elliptical crack to a through-the-thickness crack. The interaction of multiple repairs will be accounted for in the analysis. As in Task 1, the ability to analyze the effect of size, shape, thickness, tapering and material properties of the composite patch on the crack-tip stress intensity factors and adhesive shear stresses will be provided.

2.1.2.3 Task 3

Implementation of the ability to account for the effects of cold working, interference-fit fasteners and clamp-up on the crack-tip stress intensity factors of part elliptical and through-the-thickness cracks will be accomplished. This task will require that an analysis to determine the residual stress and/or strain fields produced by cold working, interference-fit fasteners and clamp-up be incorporated into the software. This analysis will be carried out by the implementation of a contact algorithm and an analytical solution for the expansion of a hole in an elastic-plastic material. Once determined, these residual stress and/or strain fields will be introduced as an applied loading in the usual Finite Element Alternating Method.

2.1.2.4 Task 4

In Task 4, the capability to handle multiple surface and corner cracks in monolithic aircraft structure will be added to the COMPAT_3D software developed in Task 1.

2.1.2.5 Task 5

In Task 5, the ability to account for thermal effects during patch application as well as aircraft usage will be implemented in the software. A composite repair is first subjected to thermal cycling during the installation process. This thermal cycling then continues during each flight of the repaired aircraft. The concern with the installation process is that during the elevated temperature curing of the adhesive, the metallic structure will expand much more than the composite patch. As a result, the repair process will produce a residual tension in the metallic structure. Obviously, this residual tension reduces the effectiveness of the repair and should be accounted for in the design. During operation of the repaired aircraft, the differences in the thermal expansion properties of the composite patch and metal structure will produce loads which interact with the mechanical loading due to gusts, maneuvers, etc. Fatigue calculations which determine inspection intervals of the repair should take these thermal loads into account.

2.1.2.6 Task 6

In Task 6, the ability to assess the effect of disbonds on the effectiveness of a composite repair will be implemented in the software. Disbonds can occur due to the development of high shear and normal (peel) stresses between the adhesive and the metallic structure. Regions near the crack and along the edges of the composite repair are particularly susceptible to disbonds. Another possible source of disbonds in composite patch repairs is impact damage. Impact damage can be caused by runway debris, damage induced by fork lifts, dropped tools and similar mishaps. No matter what the cause, disbonds between the composite patch and repaired structure increases the stress intensity factors of the repaired crack and thus reduces the effectiveness of the composite repair. Therefore, for a more realistic estimation of the residual strength and remaining life of the repaired structure, disbonds should be considered.

2.1.3 COMPAT_3D Example Application
The FEAM-based methodology implemented in COMPAT_3D is demonstrated by analyzing a generic composite patch repair typical of those made to the Macchi and Mirage III main landing gear wheels [9]. The problem is idealized as a semi-elliptical surface crack centrally located in a rectangular block. Fig. 6 shows one quarter of the problem modeled. COMPAT_3D was used to calculate the stress intensity factors along the crack front in both the repaired and unrepaired structure. Table 2 compares the stress intensity factors calculated with COMPAT_3D with those given in [9] (point *d* is the point of deepest penetration while point *s* is the point where the crack intersects the free surface). In addition, the Newman & Raju solution for the unrepaired case is given in Table 2. The agreement between these solutions is good. It is noted that the solution obtained in [9] and the Newman & Raju solution used explicit crack tip meshing, which is not necessary with COMPAT_3D. Thus, solutions of at least the same accuracy can be obtained with COMPAT_3D with only a fraction of the effort (i.e., no remeshing for different crack sizes and locations). Additional output which can be obtained from COMPAT_3D are shown in Figs. 7 and 8. Fig. 7 shows the adhesive stresses acting on the aluminum block while Fig. 8 compares the crack opening displacements of the repaired and unrepaired case. In Fig. 8, note how the crack is not modeled explicitly with finite elements.

2.2 Repair of C-141 Fuel-Transfer Weep Holes in Lower Wing Structure

The Lockheed C-141 Starlifter is a long range, heavy logistics transport. The original design was to a 30,000-hour service life using the fail-safe criteria existent in the early 1960's. The design gross weight was 318,000 pounds, with a maximum takeoff weight of 316,00 pounds and a maximum design payload of 72,131 pounds. The 'stretch' modification increased the cargo volume capacity and added an in-flight refueling capability. Changes in operating limitations increased the maximum gross weight to 325,000 pounds (higher in emergency war operations). The wings are constructed of 7075-T6 extruded aluminum. The inner-wing lower surface consists of eleven extruded panels connected in the spanwise direction with the Taper-Lok fastener system. The panels are stiffened by integral risers. To prevent fuel entrapment between these risers, 750 "weep holes" are drilled at intervals along the bases of the risers in each inner wing, Fig. 9. (The outer wings are constructed in the same manner, but the stresses are such that fatigue in the weep holes does not present a problem.)

Cracking in the weep holes occurs and propagates in normal fatigue fashion; some cracks are accelerated because of rough surface finish at the hole edges. The origin of the cracking can be at the top or the bottom of the hole. Some cracks grow from the top and bottom simultaneously, Fig. 10. Weep hole inspection has been a Programmed Depot Maintenance (PDM) requirement since 1982. Time Change Technical Order (TCTO) 1C-141-526 was conceived to inspect, ream and cold work the weep holes to remove incipient cracks.

In 1993, cracks on in-service aircraft led to an inspection of several aircraft at the depot. The inspection revealed that cracking in weep holes was widespread. A risk assessment was performed which indicated that immediate action was required. The ensuing investigation revealed that the existing inspection equipment and techniques were not adequate to detect cracks with the Probability Of Detection (POD) required. The sheer number of holes and their inaccessibility made it impossible to obtain an acceptable POD with hand-held bolt-hole-eddy-current (BHEC) probes. It became painfully obvious that cracks were being missed during inspection. A flexible automatic probe was quickly developed and tooling designed for reaming the weep holes was adapted to use as guides for the flexible probes. A plan was formulated to inspect and repair the force as soon as possible while operating as safely as possible. TCTO 1C-141-549 was issued to inspect the weep holes - all aircraft were to be inspected by the end of Dec 1993. Based on the number of flight hours, 45 aircraft were grounded and flight limitations were imposed on others. Field teams were deployed and all aircraft were inspected within three months. More than 13,000 cracks were detected. Oversizing the weep holes removed over 80% of the cracks. Wing panel replacement was required on 51 aircraft. Composite repairs were accomplished on more than 125 aircraft. After the initial inspection was completed and near normal flight operations were restored, the program to inspect, ream and cold work weep holes was resumed. With each inspection, the POD increases. The refined procedures were incorporated into TCTO 1C-141-526 and into the PDM requirements.

As previously stated, the majority of the cracks found were small enough to be removed by reaming the hole oversize. Holes could be reamed up to 0.391 inch in diameter. Any crack not removable at 0.391 inch required repair. The preferred option of repair was to install boron epoxy patches over the cracked weep hole. Composite repair of metallic structure on the C-141 had been under study since the late 1980's. The requirement for so many repairs on the lower wings made this method of repair imperative. By the summer of 1993, Wright Laboratory had developed an acceptable process for the application of composite patches inside fuel tanks. That method was provided to the Technology and Industrial Support Directorate of the Warner Robins-Air Logistics Center (WR-ALC/TI), who in turn adapted it to the industrial environment. By September of that year, composite repairs were being installed on aircraft at the Warner Robins-Air Logistics Center (WR-ALC) around the clock, seven days a week, by WR-ALC/TI and by Composite Technologies Incorporated (CTI). In the weeks and months that followed, Lockheed, Chrysler and Wright Laboratory were also installing composite repairs on the weep holes. Panel replacements were accomplished at the facilities of WR-ALC, Chrysler and Lockheed.

The basic repair configuration consists of three boron epoxy doublers - one on each side of the riser and one on the exterior surface of the wing panel (Fig. 11). It became necessary to split the external doubler whenever the installation bridged the gap between panels at spanwise splices. Crazing along the splice line occurs if this step is not taken. Surface preparation by grit blasting with aluminum oxide and silane application is required. Patches are precured in an autoclave prior to installation on the aircraft. They are applied with FM73 adhesive, vacuum bagged and heated with heating blankets. Heat is applied from the exterior surface of the wing panel. Heat transfer is sufficient to cure the riser patches inside the tank without the need to put heating blankets inside the fuel cells. The temperature is carefully monitored throughout the heating process using thermocouples. After curing, the quality of the installation is checked using thermography. Voids and inconsistencies show up as differently shaded areas on the thermographic images. A protective fiberglass layer is bonded on the outside of the installation.

There were several lessons learned in route to a smooth process. Aluminum oxide grit used in surface preparation <u>must</u> be contained. Fine powdery residue is the by-product of the blast operation. If the residue was not contained, it clogged fuel filters, got into fuel controls and caused engine roll backs. Double containment tents and meticulous post operation cleanup controlled the problem. The installation process is a continuous one. Once begun, it must be carried out to fruition. Large ambient temperature variations or precipitation will cause a stop in the process. At that point, it is necessary to begin the process all over again. Indoor, environmentally controlled facilities are <u>highly</u> recommended. As with any bonding operation, installation technicians must be trained and experienced. Thermal expansion must be taken into account both in the load direction and in the non-load direction. In the case where patches bridged spanwise splice gaps, it caused crazing in the epoxy between the boron strands. Several repairs were accomplished before this phenomenon was discovered. The solution was to cut the patches along the gap line. Later designs incorporated the four patch configuration in these areas. When it becomes necessary to remove an installed doubler, the only effective method found to date is to grind off the patch. Naturally, great care must be taken to protect the aircraft structure in that situation.

When the number and/or proximity of cracks were such that too many composite patches in a relatively small area were required, replacement of a wing panel was necessary. Some aircraft required several panel replacements.

The initial inspection of weep holes was completed in 1993. Most aircraft have had multiple inspections to increase the POD. All weep holes have been reamed to an oversize dimension to remove incipient cracks and cold worked to retard fatigue crack initiation. Each aircraft receives a weep hole inspection at PDM. Each repair location is inspected regularly for repair integrity and for potential crack growth. External patches are inspected yearly at the Isochronal inspection. External and internal patches are inspected at each PDM. Eddy current surface scan of the wing panels around the patch peripheries is also accomplished at these times.

In summary, a complicated situation occurred in May 1993. All existing data indicated that major multi-site, multi-element damage in the weep holes of the inner wing lower surface was present. It appeared that all aircraft might be affected and that flight restrictions must be imposed. There was not a reliable NDI technique available for the inspections, and even if inspections were possible, there was no in situ technique available to rapidly repair all the affected aircraft. Furthermore, there were no replacement wing panels in supply. By August, the NDI personnel in WR-ALC/TI, working with a contractor, had developed a viable technique and designed and built the necessary equipment to accomplish the inspections. In August, personnel from Wright Laboratory demonstrated the installation of composite repairs on the inner wing lower surface. By September, inspections began on the fleet. This was accomplished by contract field teams, WR-ALC and contract maintenance facilities. Composite repairs began immediately. They were accomplished by WR-ALC/TI and CTI at Robins Air Force Base, by Lockheed in Marietta, by Chrysler in Waco and by Wright Laboratory at Wright-Patterson Air Force Base. Inspections were completed in December 1993. CC Industries began delivery of panels in February 1994 and panel replacement began shortly thereafter. Panel replacements were accomplished at WR-ALC, Chrysler, and Lockheed. All repairs were completed. This is a perfect example of teamwork between separate government organizations [WR-ALC/TI, C-141 System Program Office (WR-ALC/LJ), Defense Logistics Agency (DLA), Wright Laboratory and operating units] and contractors (Chrysler, CC Industries, CTI, and Lockheed).

2.3 Bonded Repairs of Fatigue-Cracked Fuselages

A damage-tolerant approach discussed in this section considers the application of the unidirectionally reinforced glass fiber/aluminum laminate GLARE® 2 to the bonded crack patching of fatigue-damaged fuselage structures. Most bonded repairs to date have worked successfully on small areas of thick, fatigue-cracked structures using boron/epoxy composites. Extending the lives of aging transport fuselage structures, however, may involve repairs to thin fuselage skins and lap joints, which see maximum structural loads at very low operating temperatures. The large areas potentially affected may require operators to search for high-performance, yet more affordable patch materials.

The key elements of bonded repairs to cracked structures can be summarized in five main areas:

- the patch must reduce the stress intensity factor (K) at the repaired crack tip enough to greatly slow or stop crack growth,
- the patch must not increase stresses in the repaired skin, adjacent to the patch, to the point where new fatigue cracks occur,

- the normal stresses in the patch must not be too high,
- the adhesive shear strains near the crack must remain reasonably low, and
- the peel stresses in the bond line must not cause delamination of the patch.

Parametric studies of crack patching by Fredell *et al.* [10] predicted that unidirectionally reinforced GLARE® 2 should out-perform boron in crack patching of pressurized fuselage structures. This stems from the large relative difference in thermal expansion coefficients between boron and aluminum. The results of these studies show thermal properties have a greater influence on crack patching efficiency (i.e., reduction of stress intensity factors) than the elastic modulus of the patch. Large differences in thermal expansion also cause high adhesive shear strains at low temperatures, which can adversely affect the durability of bonded repairs. (GLARE® patches are more thermally compatible with aluminum than boron.) This effect is significant only in cases where low temperatures coincide with high flight loads.

When a bonded repair is placed on an aircraft, the structure is heated only very locally. The surrounding, cooler structure acts to restrain the free expansion of the heated area, but the patch is free to expand, as explained by Rose [11]. A patch with a moderate coefficient of thermal expansion (CTE), like GLARE® 2, will expand more than the restrained cracked structure. Hence, after adhesive curing and the initial cooling step to room temperature, the moderate CTE patch will be in residual tension, and the crack face will be compressed.

On the other hand, low CTE patches like boron expand about the same amount as the constrained structure being repaired. Thus, upon cooling to room temperature, the patch, bond line and crack are relatively stress-free. This explains the good service experience with boron patches on fighter aircraft, which experience maximum maneuver loads at moderate temperatures. However, pressurized transport fuselages attain their highest skin stresses at cruise altitude, where the outside air temperature can be as low as -60°C.

In the second cooling step (to cruise altitude), the fuselage cools uniformly and contracts freely. Now, the GLARE® patch contracts slightly less than the fuselage, leaving the crack face approximately stress-free. However, the boron patch contracts much less than the fuselage, effectively opening the crack and diminishing patching effectiveness. This second step occurs with every flight.

When modeling crack patching, if thermal expansion differences between the patch and substrate are ignored, the very stiff boron patches appear most efficient. However, when the thermal effects of cruise altitude are considered, the situation changes drastically, as shown by a comparison of stress intensity factor reduction (Fig. 12). The moderate thermal expansion coefficient of GLARE® enables it to perform much more effectively than any thickness of boron patch.

In addition to a choice of patch materials, the repairer also may specify the cure temperature during patch installation. When repairs are performed on or near structures containing absorbed moisture (e.g., honeycomb), cure temperatures under 100°C are desired. Furthermore, a lower cure temperature can reduce thermal buckling problems. Otherwise, cure temperatures are set by equipment limitations. Temperatures around 120°C are used with toughened epoxies to gain good bonds in reasonable amounts of time. When materials of different thermal expansion coefficients are bonded, cure temperature can affect residual stress states as well. Baker [12] recommends curing at "the lowest possible temperature" to minimize residual thermal stress.

Fig. 13 shows the effect of various cure temperatures on the patching effectiveness of GLARE® 2 and boron/epoxy. The GLARE ® patch benefits from a higher effective thermal expansion because of the restraint on the stiffened fuselage. In the analyzed case, the effective expansion coefficient of the stiffened fuselage is approximately equal to that of boron. Thus, in fuselage skin repairs, that of boron. Thus, in fuselage skin repairs, boron neither benefits nor suffers from a change in the cure temperature. This is because the large thermal effects with boron occur only upon cooling from room to cruise temperature.

Thermal properties affect more than the reduction of the stress intensity factor, of course. The four remaining criteria of bonded design (normal stresses in the repaired skin, normal stresses in the repair patch, adhesive shear strains, and peel stresses) must remain in the acceptable range.

Of these, the design factor probably most sensitive to patch material selection is the maximum shear strain in the adhesive bond line. A conservative design practice keeps the adhesive shear strains below one-half of yield for good bond durability. The very stiff, low-CTE boron patches suffer from high adhesive shear strains. This may be attributed to the large mismatches in both thermal expansion and elastic modulus properties when compared with the aluminum skin being repaired. With GLARE ® patches, the shear strains are much lower.

As discussed earlier, the performance of boron patches is insensitive to cure temperature. Consequently, the high adhesive shear strains experienced with boron patches cannot be reduced significantly by curing at a lower temperature. Fig. 14 illustrates how small the effect of reduced cure temperature is on shear strains in the boron patch bond lines. With GLARE® 2, the effect is reversed: a higher cure temperature (100 to 120°C) actually benefits the bond by reducing adhesive shear strains at operating temperatures. From these results, one can infer that only the second temperature step (room to cruise temperature) is critical for low CTE patch materials like boron.

The results of constant amplitude fatigue tests performed on bonded repairs to a large flat stiffened panel and a pressurized unstiffened fuselage "barrel" were presented in a separate paper by Fredell *et al.* [13]. These specimens simulated pressurized transport fuselage structures with multiple-site fatigue damage. Bonded GLARE® 2 patches successfully halted the growth of repaired multiple fuselage fatigue cracks. Furthermore, the presence of unrepaired fatigue damage adjacent to the bonded patch in the stiffened flat panel did not compromise the damage-tolerant nature of the repair. Multiple unrepaired small cracks near the repair grew very slowly, allowing relaxed structural inspection intervals.

Analysis, detailed design studies and extensive testing by Fredell *et al.* [10, 13, 14] have shown that GLARE® 2 patches may be employed successfully in fuselage crack patching. Use of a bonded GLARE® 2 patch with an extensional stiffness roughly equal to the damaged skin produces the following results:

- stress intensity factor reductions of 90 to 100 percent,
- low stress concentrations in the repaired skins near the patch,
- normal stresses in the patch below one-half the patch yield stress, and
- peak adhesive shear strains below one-half the yield strain.

The final element, peel stresses in the bond line, can be controlled adequately by long patch overlaps and thickness tapering at the tips of the patch.

The bonded GLARE® 2 patch accomplishes its intended function---enhancing structural durability by stopping the crack---without causing new difficulties in the existing structure. Low adhesive shear strains and bond-line peel stresses, combined with good surface preparation and bonding practices, ensure good bond durability and delamination resistance. Designing for moderate stresses in the patch and the choice of a fatigue-resistant patch material like GLARE® 2 avoid fatigue damage in the patch itself. The result is durable, damage-tolerant repairs that allow operators to extend the useful structural lives of their aging fleets economically and with confidence.

In summary, high-strength high-modulus composites like boron/epoxy are useful in many crack-patching repair applications. However, the unique combination of stress fuselage repair require a patch material with a better thermal compatibility. Analysis and testing have shown the fiber-metal laminate GLARE® 2 to be a better choice for pressurized fuselage crack patching. In addition, adhesively bonded GLARE® 2 patches demonstrated their damage tolerance capability by reducing the growth of nearby multiple-site damage.

To ensure the continued safety of aging pressurized transport aircraft, operators and maintainers must adopt a new, damage-tolerant method of fuselage structural repair, as discussed above.

For bonded repair of intact fuselage fatigue cracks, the unidirectionally reinforced GLARE® 2 laminate is a patch material that satisfies all design requirements for durability and damage tolerance. GLARE® 2 surpasses the performance of higher strength, very stiff boron/epoxy composites in pressurized fuselage repairs because of complex thermal interactions. Extensive analysis and testing of this concept enable transport aircraft operators to deal effectively with multiple-site fatigue damage.

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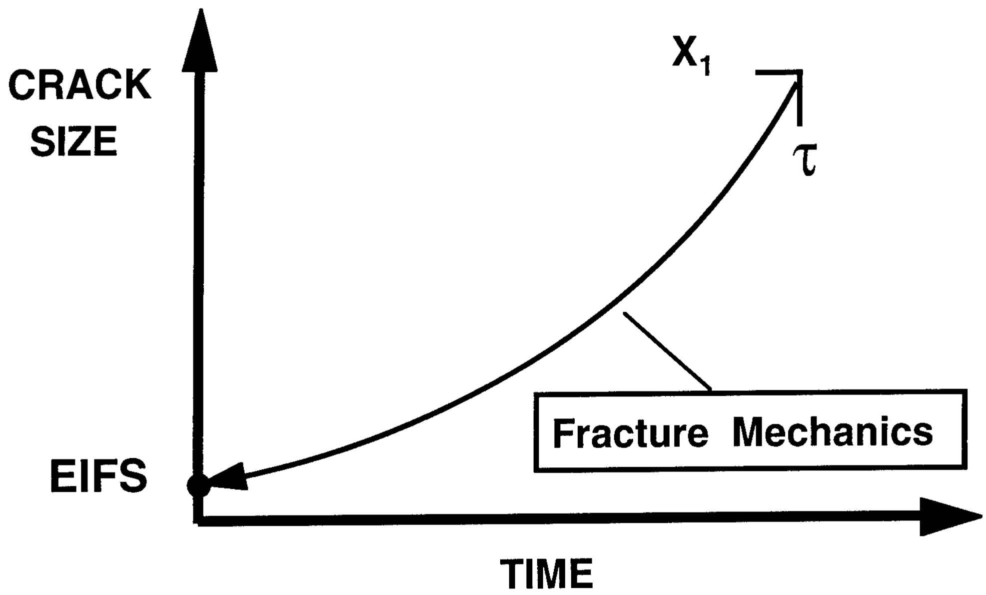

Figure 1. Equivalent Initial Flaw Size

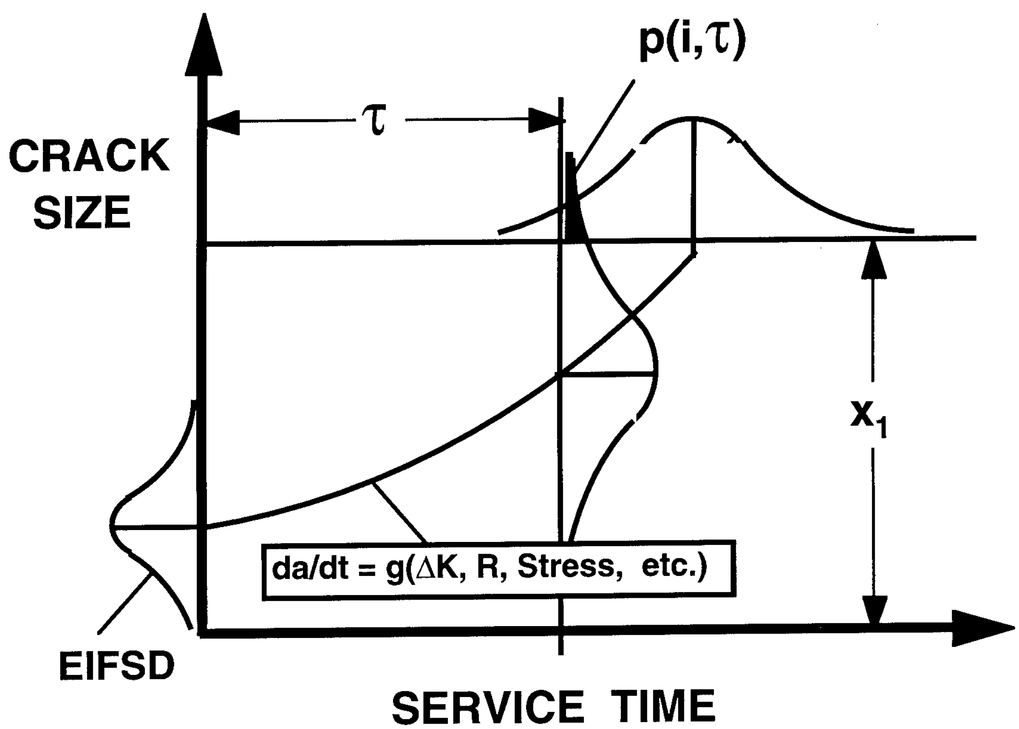

Figure 2. EIFS Approach in Probabilistic Framework

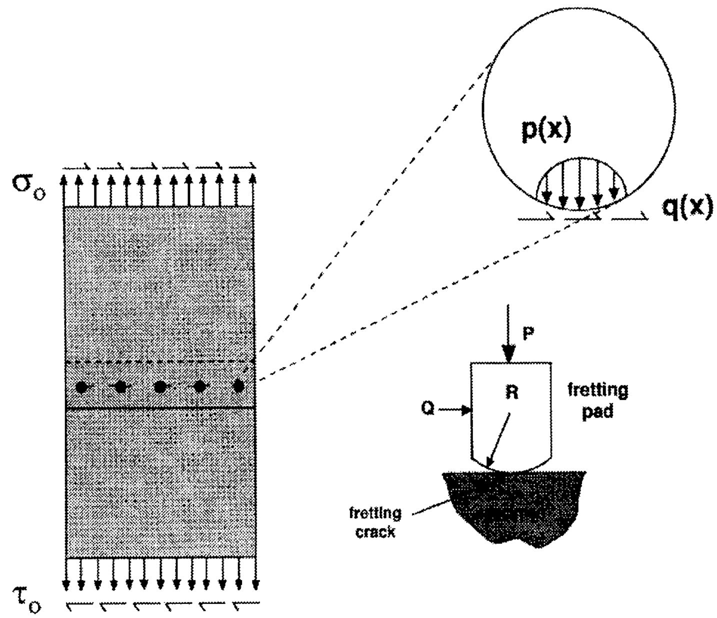

Figure 3. Schematic Representation of the Relationship Between an Aircraft Structural Lap Joint and the Fretting Contact Problem

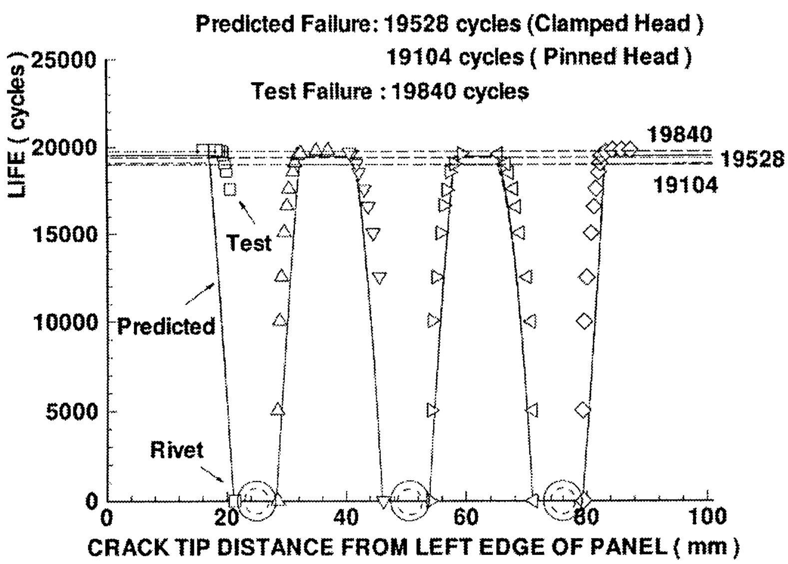

Figure 4. Comparison Between Experimental and Predicted Fatigue Crack Growth Behavior in a Lap Joint Specimen Which Contains One Row of Precracked Fastener Holes.

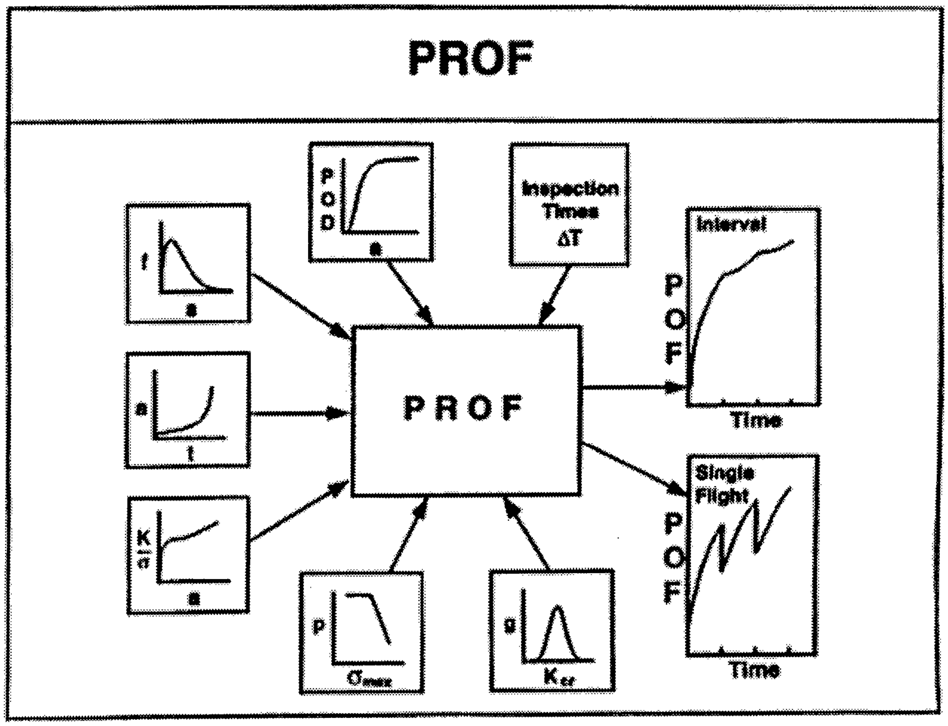

Figure 5. Schematic of Probability Of Fracture (PROF)

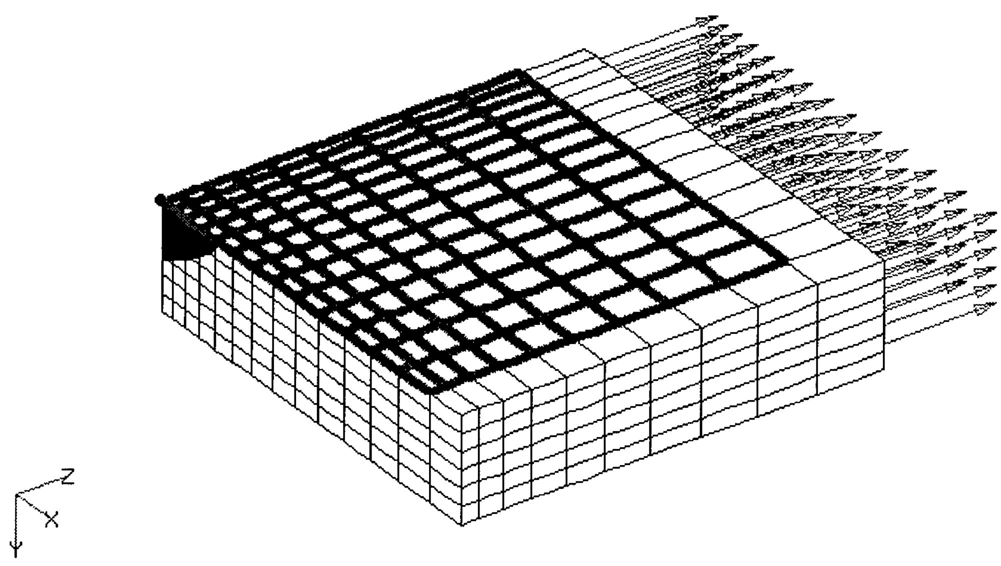

Figure 6. Repair of Surface Flaw (1/4 of Structure Modeled)

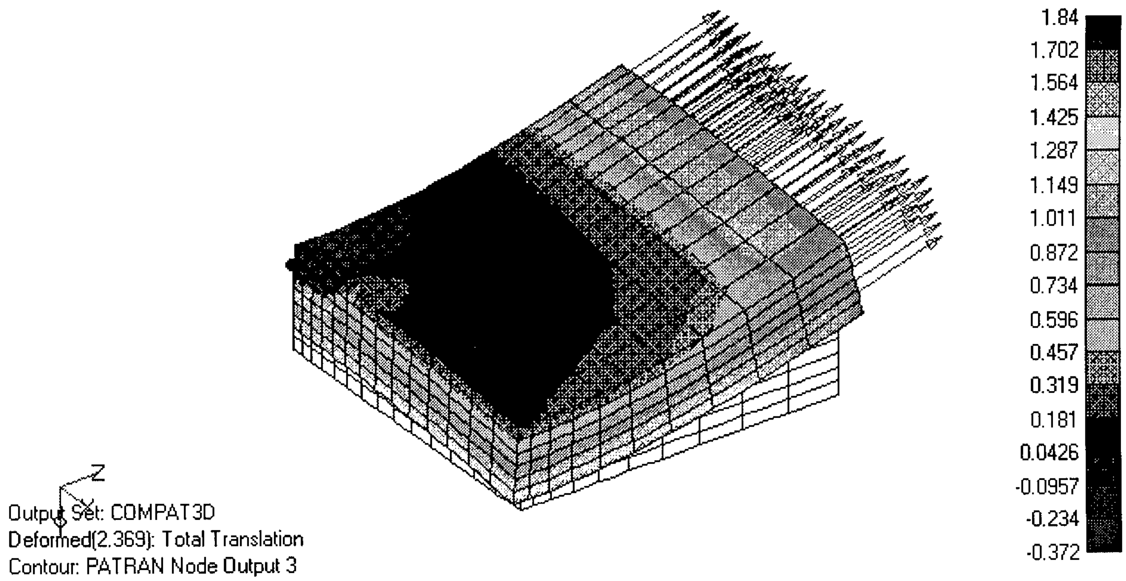

Figure 7. Adhesive Stresses Acting on Aluminum Block

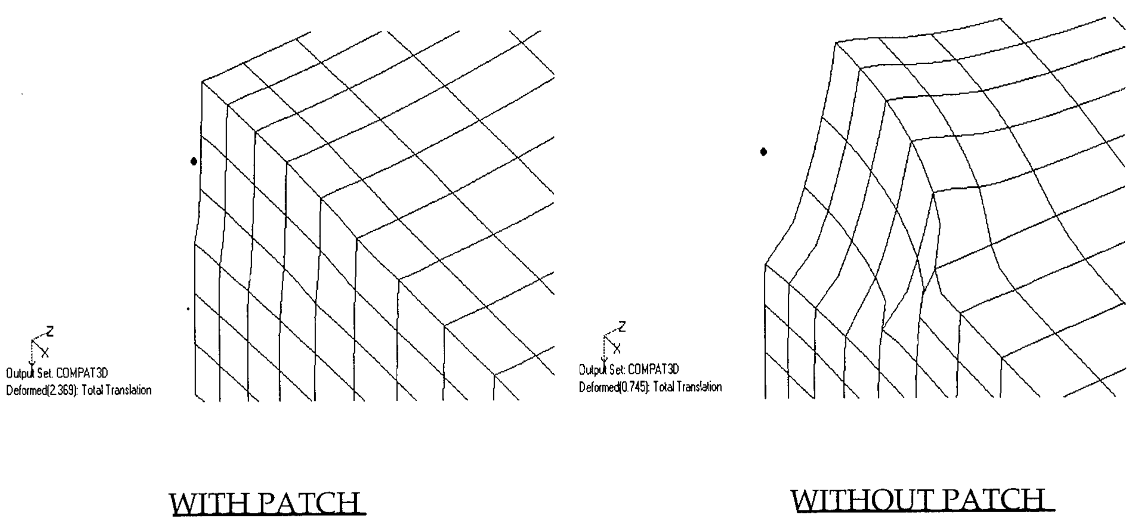

Figure 8. Repaired and Unrepaired Crack Opening Displacements

- WINGS MADE FROM INTEGRALLY STIFFENED 7075-T6 EXTRUSIONS
- HOLES DRILLED AT BASE OF RISERS TO PREVENT FUEL ENTRAPMENT (750 PER INNER WING)

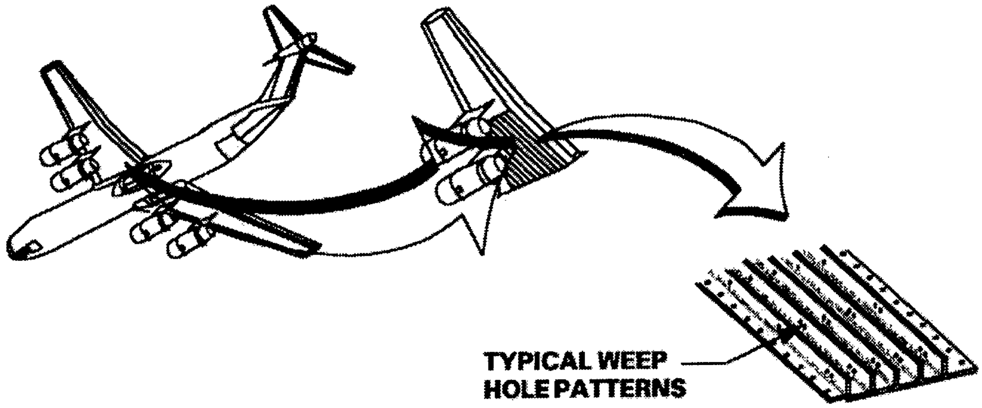

Figure 9. C-141 Weep Hole Description

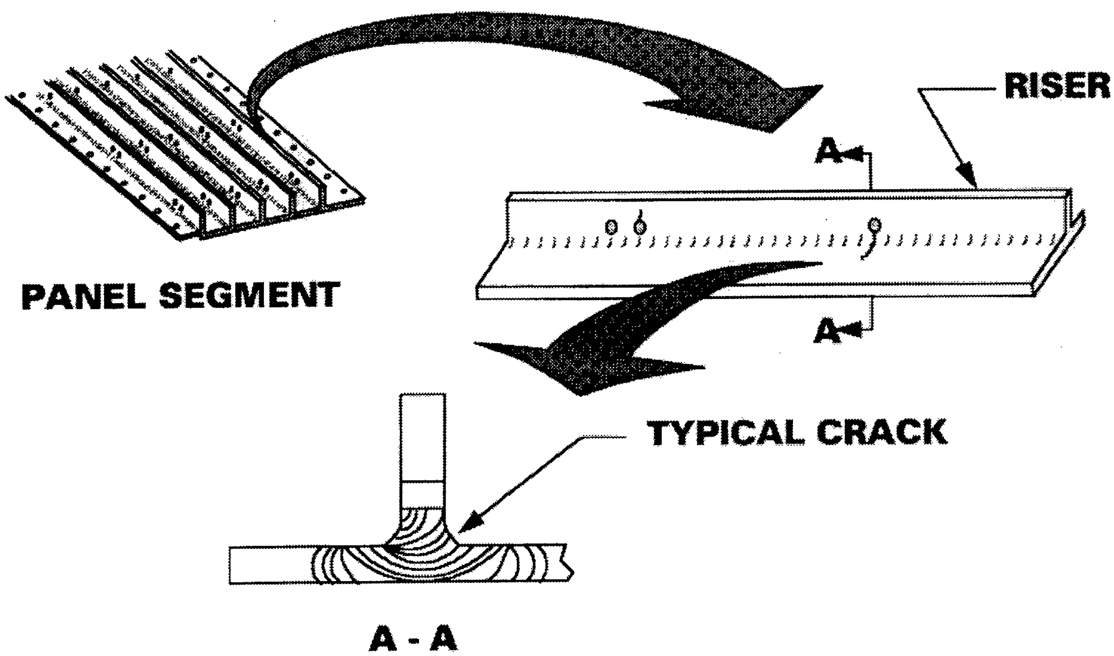

Figure 10. Typical C-141 Weep Hole Cracking

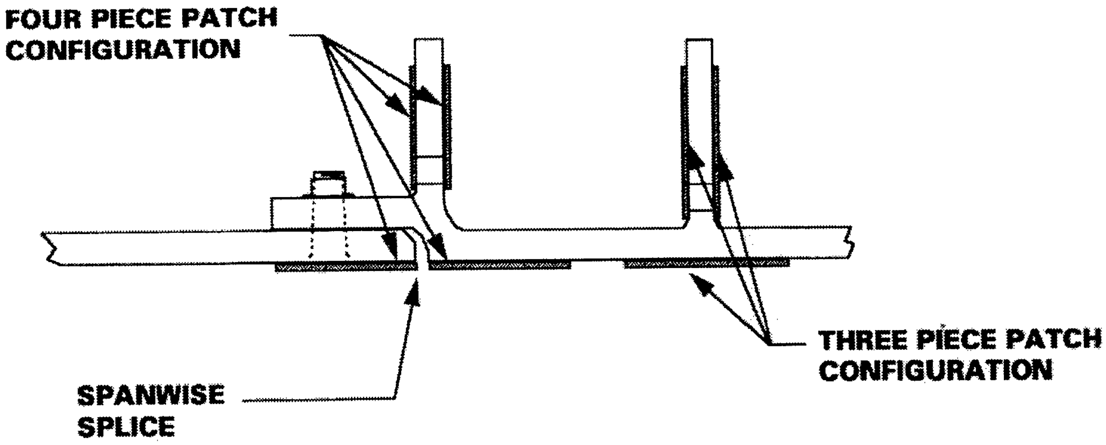

Figure 11. Typical Composite Weep Hole Repairs

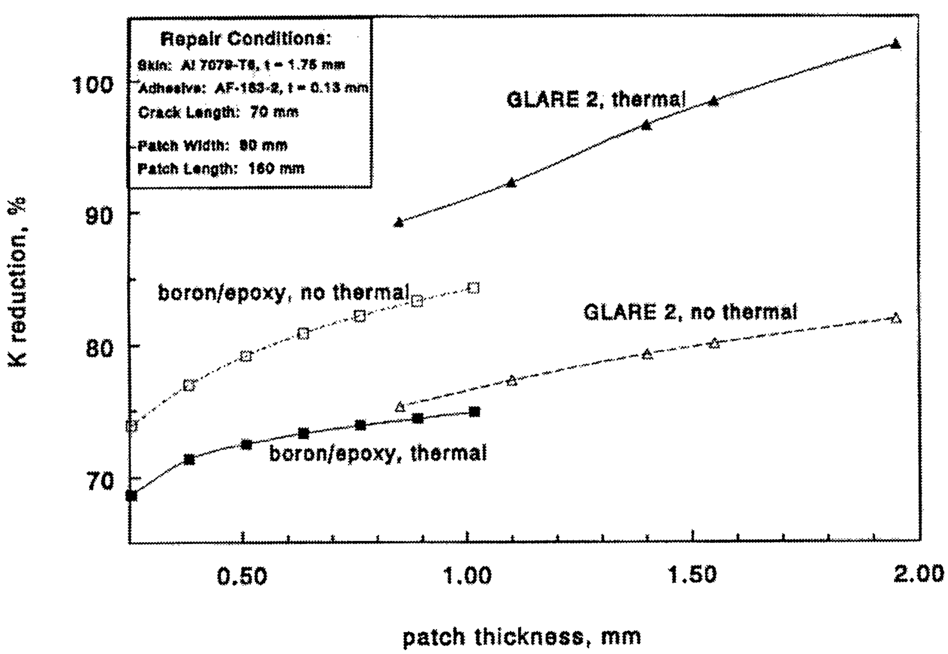

Figure 12. Comparison of Reduction in Stress Intensity Factor K for Bonded GLARE® 2 and Boron/Epoxy Patches, With and Without Thermal Effects.

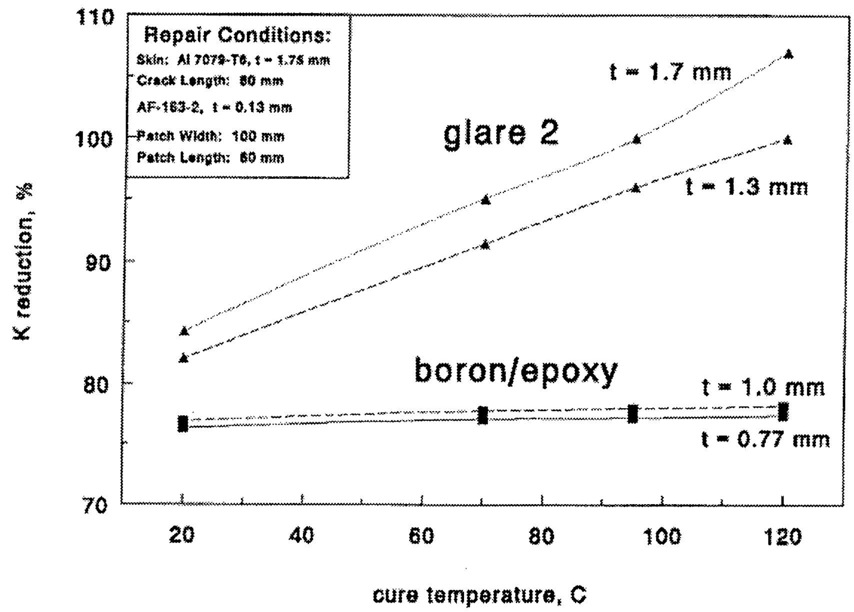

Figure 13. The Variation in Fuselage Crack Patching Effectiveness With Cure Temperature.

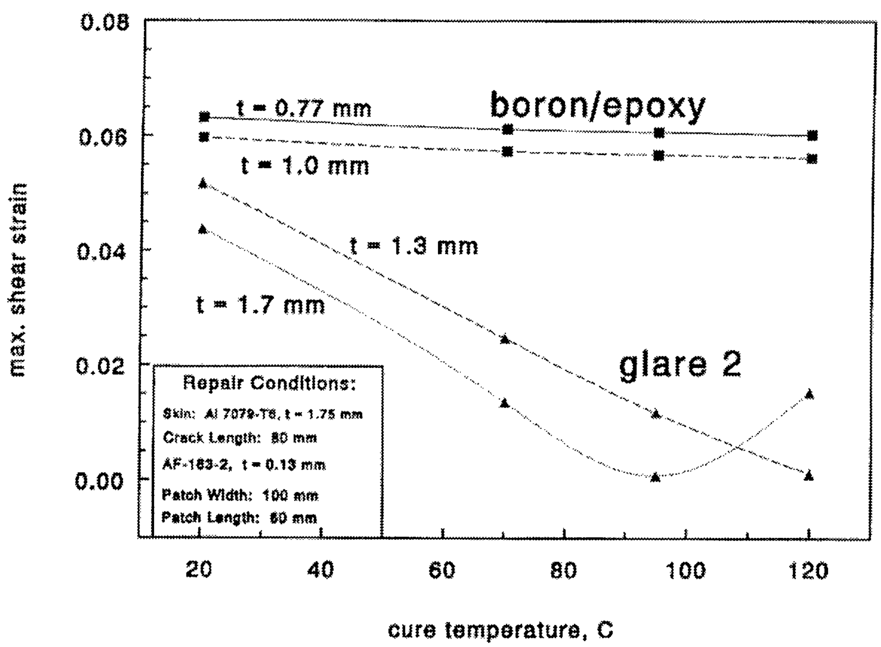

Figure 14. The Influence of Curing Temperature on Maximum Adhesive Shear Strain.

Table 1. Summary of Applications of EIFS Approach and Information Provided

EIFS APPROACH APPLICATION	INFORMATION PROVIDED
DURABILITY ANALYSIS	• Probability of crack exceedance, $p(i,\tau)$ • Extent of damage (No. of details $\geq x_1$ at any service time τ) • Damage accumulation rate • Economic life
RISK ASSESSMENT	• Probability of failure (i.e., exceeding a_{cr}) at service time τ • Risk rate (probability of failure per flight) • Probability of exceeding the functional impairment crack size, a_f, at service time τ
MULTI-SITE DAMAGE (MSD) AND WIDESPREAD FATIGUE DAMAGE (WFD)	• No. of structural details expected to exceed specified crack size limits (e.g., x_1, and a_f) at any service time τ • Damage accumulation rate • Ligament breakage
LIFE EXTENSION OF AGING AIRCRAFT	• Risk rate (probability of failure per flight at service time τ and time $\tau + \Delta\tau$) • Probability of crack exceedance at service time τ and at $\tau + \Delta\tau$ • Extent of damage at service time τ and at $\tau + \Delta\tau$ • Damage accumulation rate between service time τ and $\tau + \Delta\tau$ • Inspection interval (when to inspect) • Decision for repair and replacement • Aircraft utilization tradeoff options
RELIABILITY-CENTERED MAINTENANCE	• Probability of a Class A mishap per flight hour • Initial inspection • Re-inspection interval

Table 1. Summary of Applications of EIFS Approach and Information Provided (Cont'd)

EIFS APPROACH APPLICATION	INFORMATION PROVIDED
RELIABILITY-CENTERED MAINTENANCE (cont'd)	• Maintenance man-hours per flight hour (MMH/FH) • Inspection window based on allowable risk rate • Inspection technique options and tradeoffs
MAINTAINABILITY AND SUPPORTABILITY	• No. of structural details expected to exceed specified crack size limits at any service time τ • Expected No. of details requiring inspection and repair maintenance • Expected range of crack size at any service time τ • Inspection technique options and trade-offs • Maintenance man-hours per flight hour (MMH/FH) • Estimation of material, tooling, spares, support equipment skill level, and manpower requirments to support maintenance and repair schedule
S-N CURVES	• Estimate S-N curve trajectories for selected probabilities, different K_t, R-ratio, reference crack size, etc.
EFFECT OF MATERIAL QUALITY ON FATIGUE PERFORMANCE	• Effect of microstructure (e.g., voids porosity, grain size, etc.), yield strength, etc., on material fatigue performance
EFFECT OF MFG. QUALITY ON FATIGUE PERFORMANCE	• Effect of drilling, cold-working, etc., on fatigue performance of critical details

Table 2. Mechanical, Physical Properties of 2024-T3 and GLARE® 3

MATERIAL	LAY-UP	THICKNESS (mm)	E_{11}, E_{22} (GPa)	$\sigma_{0.2}$ (MPa)	$\sigma_{bluntnotch}$ (MPa)
2024-T3	monolithic	variable	72	359	440
GLARE 3	2/1-0.3	0.85	60	315	452
GLARE 3	3/2-0.2	1.1	57	295	469
GLARE 3	3/2-0.3	1.4	58	305	456

Airframe Inspection Reliability

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1. SUMMARY

Inspection reliability is a cornerstone of the damage tolerance philosophy underlying the U.S. Air Force Airframe Structural Integrity Program (ASIP) and their Engine Structural Integrity Program (ENSIP), which are designed to ensure continued airworthiness of its fleet. Inspection data are essential to deriving inspection thresholds and inspection intervals—elements of every maintenance program for the constituents within a fleet. Frequency and the method of inspection are primary drivers of maintenance costs and thus life-cycle costs. On the other hand, structural safety also depends on inspection reliability, i.e., the ability to detect damage in a timely fashion. This presentation highlights certain aspects of inspection reliability that primarily relate to airframe structures and recommends the establishment of an international data collection and cataloging activity for improving inspection reliability.

2. INTRODUCTION

Aircraft structures are designed and built to withstand all credible loads that can be encountered during service. Typically, their strength is qualified in pristine condition through analysis, subscale model testing, and full-scale tests. As they age, degradation is inevitable due to fatigue or corrosion for instance. Hence, a certain amount of strength loss due to degradation is factored into the design. Further substantiation of their continued airworthiness is made necessary to preclude the possibility of premature failure brought about by any one of several reasons as depicted in Figure 1 [1]. The effects of potential battle damage and damage due to other external sources also need to be guarded against. Invariably one of the following options is chosen to guarantee continued airworthiness [2].

a. Periodic inspection.
b. Safe-life design.
c. Periodic proof testing

According to U.S. Air Force practice [3], the method of periodic inspection is clearly preferred to assure structural integrity during the operational life of an aircraft; only exceptional circumstances, as discussed in [2], may require adoption of one of the other methods. This presentation is about inspection reliability—a cornerstone of damage tolerance design and assurance of continued airworthiness through periodic inspection and maintenance.

3. ROLE OF INSPECTION AND INSPECTION RELIABILITY

Inspection tasks involve looking for cracks and corrosion in a variety of components, as indicated by Figure 2. Since fatigue crack growth material properties, stress levels in the vicinity of a crack, and the fracture resistance of the structural part differ widely, different components invariably can tolerate differing extents of damage, be they due to fatigue cracks, accidental damage, or corrosion. Many components can tolerate considerable damage prior to failure, as illustrated in Table 1.

The data in the table are derived from the Service Difficulty Reporting System (SDRS) [4]. The SDRS is a repository of information pertaining to mechanical discrepancies encountered by civil aircraft in commercial service. SDRS is maintained by the Federal Aviation Administration (FAA). During construction of the table, when the SDRS records for the period 1990 to-date were scanned, there were some 40,000 instances of fatigue cracks being discovered. During 1991, 27 records were encountered, each indicating detection of a crack at least ten inches in length—all of which goes to show that the occurrence of cracks in aircraft structures is quite common, and it should not cause undue alarm as long as the maintenance program ensures their detection before failure occurs. However, it should not be construed that all flight-safety-critical components are as tolerant of cracks as long as Table 1 might suggest. The key to preventing failures of course is to adjust the inspection frequency, based on the inspection reliability, to make certain that cracks are detected in time.

4. CONSTRUCTION OF THE PoD CURVE

The sensitivity and reliability of inspection is best characterized by a Probability of Detection (PoD) curve. In its simplest form, the inspection probability is calculated by dividing the number of cracks detected, n, by the number of opportunities, N. This scheme provides a single value and does not account for the differing crack lengths in the population. If cracks are grouped according to size and the inspection probability, (n/N), calculated for each size, a histogram can be constructed as shown in Figure 3 [5]. To establish statistical validity, a typical PoD curve, as illustrated in Figure 4, has to be constructed from a large aggregate of data about successful crack detection and nondetection. Some issues that are related to the difficulty of having a sufficiently large data set to construct a PoD curve are discussed later.

5. USE OF THE PoD CURVE

The United States Air Force (USAF) uses the 90% probability of detection (PoD) and 95% confidence level as the criteria for setting inspection schedules [3] as follows:

a. From the PoD curve that characterizes the inspection method, the crack length corresponding to 0.9 probability is read off (Figure 4). The length is termed as the detectable flaw size, a_d.

b. From the crack length vs (pseudo) time curve constructed from the material property data (fatigue crack growth rate property) and stress analysis results, the time corresponding to a_d, T_d, and the time when the crack will go critical, T_c, are read off (Figure 5).

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c. An assumption is made about a potential initial defect (rogue flaw). The time corresponding to this size is of course equal to zero.

d. The first inspection is set at time $(T_c)/2$; subsequent inspections occur at intervals of $(T_c - T_d)/2$—that is, the second inspection is scheduled at $(T_c - T_d/2)$ and the last opportunity for detecting the crack will be at $(T_c + T_d)/2$. Thus, there will be at least one opportunity to detect a crack after the first inspection prior to it going critical but typically there will be two opportunities.

It is evident from the above stated that the inspection option is implementable only if T_c is larger than T_d.

Table 2, which has been drawn from Reference 6, shows a variant scheme to calculate, from the PoD curve and an assumed inspection interval, the time when the cumulative probability of detection will reach 0.90. The crack length corresponding to the latter time may then be compared against the critical crack length, a_c. Table 3, also from Reference 6, shows that even a small increase in the detection performance of the inspection system will allow significantly longer inspection interval which, in turn, can provide significant relief in inspection costs. The same point applies to USAF practice; viz., the shorter the detectable flaw size the shorter is T_d and the longer is the inspection interval $(T_c - T_d)/2$.

Figure 6, showing a sketch of the four possible outcomes during inspection of a structure, illustrates the same point differently. The upper-left and the lower-right outcomes, shown in the figure, are positive attributes of inspection reliability whereas the lower-left and the upper-right outcomes are undesirable. Thus, the question of how small a crack can be detected becomes less important than how large a crack can consistently be found. Secondly, an indication of a crack when none is present—false calls—can be quite penalizing from the economic point of view. It follows that the ideal system response when inspecting should be as sketched in Figure 7.

6. SOME FACTORS PERTAINING TO INSPECTION RELIABILITY

One of the early, substantial efforts to quantify inspection reliability is described in Reference 7. Structural components that used to belong to in-service aircraft were subjected to inspection, by several maintenance inspectors at USAF logistics centers, using traditional nondestructive inspection (NDI) methods, and their responses were recorded. The results of the study are limited, however, because it did not take into account several important factors that can influence the results. For instance, no distinction was accorded to the fact that not all components were made of the same material or the same geometry. On intuitive grounds and more recent modeling efforts, it can be shown that the signal from the NDI sensor is affected by the structure's overall geometry as it would be by a crack in that structure; likewise, the material of the structure affects the NDI response. On the other hand, if the results were to be segregated on the basis of structural shape, material, etc., one faces the dilemma of not having enough, otherwise identical, samples, many without cracks, and many with cracks of various lengths—thus, not having enough data points to construct a statistically meaningful PoD curve.

Only recently has the subject of human factors become prominent, and it has been receiving considerable attention on account of its relationship with human performance as it relates to engineering and maintenance functions. Reference 2 discusses the aspects of access and specificity in some detail.

Easy access to a location that needs to be inspected is conducive to proper viewing, or positioning of an NDI probe, to discern a crack and record the finding. On the other hand, inspection reliability can suffer in areas that have tight access. Thus, ideally, if PoD data associated with a specific part is to be acquired, the setup should faithfully recreate the access by inspectors with their NDI probes. However, that may be quite impractical because a great number of samples are needed to ensure statistical validity.

Specificity is used in the sense of whether the inspectors are looking for cracks at a specific location(s) or whether they have the task of inspecting a great many fasteners with the prospect of finding hardly any cracks among them. Obviously, boredom can play a part in degrading the ability to find an isolated and occasional crack.

The latter aspect was investigated in a study that used simulated cracks and real inspectors [8]. Both NDI equipment reliability and human performance associated with the inspection task affect inspection reliability, but it is not possible to split the convolution of the two factors into the individual parts. The study in [8] was able to isolate the influences of human factors since the simulation created "perfect" NDI equipment reliability.

The test specimen was a computer aided design (CAD) tablet with a cover plate constructed to simulate a lap splice with three rows of counter-sunk rivets in a fuselage structure. The CAD tablet was connected to a computer which could be programmed to present to the inspector a very large number of lap splices with different combinations of cracks. Inspector responses, which involved use of eddy-current probes, were measured electronically and video monitored. At the end of the six-day test period the inspectors marked their perceptions about the importance of each environmental factor on their ability to detect a crack. Scores assigned by the inspectors, on a 1-5 scale, against the factors are shown in Table 4.

Analysis of recorded inspector responses revealed that painted surfaces rated high among the reasons for degraded inspector response. It must be noted however that the overall error rates were very low. A significant effort [9], sponsored by the FAA, to measure inspection reliability involved presenting airline inspectors with structural assemblies having artificially created cracks that resembled fuselage lap splices. The test protocol, involving eddy-current inspection, did attempt to address the various aforementioned factors that can affect inspector performance; however, some questions about whether the results can be treated as representative of actual aircraft inspections at a depot remain. In any case, the results may not apply to other structural details, the wing spar for instance.

An altogether different approach to measure inspection reliability, as described in [10], was taken by a Japanese team consisting of personnel belonging to the civil aviation regulatory authority, three Japanese-flag air carriers, and a research institution. Details about visual findings of cracks in fuselage sections of Boeing-747 aircraft during routine maintenance were collected over a three-year period and analyzed. An extension of the idea contained in [10] is proposed in Reference 11.

Reference 11 advocates the undertaking of a multinational collaborative data gathering and analysis effort under the auspices of AGARD. The purposes of the effort would be to gather inspection reliability data that are naturally obtained under field conditions and, through fatigue crack growth modeling and analysis supplemented by fractographic analysis, to define the probability of detection of fatigue cracks that occur in airframes by aircraft maintenance personnel. The results from the effort would be extremely beneficial to the maintenance departments of participants in the study. They would signify how the reliability of their inspection program can be improved in a cost-effective manner. For instance, if there is a large spread in the inspection reliability data from a certain depot, analysis of the test conditions and the inhomogeneous characteristics of the inspector group may indicate the reason(s) for the scatter in the performance of the work force, making appropriate calibration of the maintenance program possible.

A multinational effort is necessary because there is virtually no inspection reliability data that reflect field conditions and which adequately document such findings. In particular the size/extent of a crack when it was found, how it was found, and what access problems were associated with it being found or not found during previous inspections are not generally reported. From what little data exists, in many cases it is impossible to discern the age of the aircraft when the crack was found and the history of the crack that escaped previous detection. Moreover, assessment of the reliability of repetitive inspection has been highlighted as one of the critical outstanding issues that is closely related to the problem of widespread fatigue cracking in aging aircraft. It is primarily due to lack of pertinent information.

The proposed cooperative project will consist of two closely related tasks:

a. Collect and catalog information related to findings of fatigue cracks in flight-safety-critical components during routine and nonroutine inspections by air force maintenance personnel.

b. Analyze the crack related information, using principles of fatigue and fracture mechanics, to derive the crack growth histories associated with each crack and thereby estimate the number of detect and nondetect events, the probabilities of detection, and the associated confidence level as a function of the variables involved in aircraft inspection.

Support for the basic idea in [11] comes from Dr. D. Bruce of DRA, Farnborough in [12] who makes the case that greater confidence in NDI reliability would assist fleet life-extension efforts. Reference 12 also cautions that careful thought is needed to establish the method of analysis, interpretation, and specifications.

At the time of this writing, it is envisioned that an AGARD-sponsored workshop will be organized primarily to address issues raised in [12]. Issues pertaining to reliability of inspection of aircraft engines are generically similar to those that surround airframe inspection reliability, but there are significant differences as well; see References 5 and 13.

In conclusion, there is a prospect of rich rewards through establishing a quantitative basis for inspection reliability and utilizing the data when developing a maintenance program. Much work needs to be done to bring it about. A multinational effort will be the best course to achieve that end.

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Table 1. Examples of Findings of Fatigue Cracks in In-Service, Commercial Airplanes

Case	Field Office Identification Number	Region Code	Date of Occurrence	Aircraft Group	Findings
1	GL019071850	WP	10/16/89	DC9	During periodic C-check, a 10-inch crack was found on the rear spar, upper cap, right wing at XCW 20.00. Stop drilled crack and installed repair per R&D 91-31620.
2	SO119072375	NM	11/27/89	B757	During special inspection, found 13-inch longitudinal fuselage skin crack adjacent to the aft VHF antenna at FS 1473. Repaired per SB 757-53A0052, part 3 repair procedure. Cycles 9,969.
3	SO39072828	NM	12/5/89	B747	Radio rack support cracked from floor line to 18 inches above floor station 440, LBL 20 at main electronics compartment. Radio rack support post cracked from floor line to 18 inches above floor at station 440, LBL 60 at main electronics compartment. Repaired as per ERA B53-10-128.
4	EA199073053	NM	12/22/89	B727	During C-8 check visual inspection, found a 10-inch crack including a 2-inch-diameter hole in the rear pressure bulkhead at body station 1183. Maintenance repaired per EA 6838.
5	EA199076208	NM	12/22/89	B727	During C-8 check visual inspection, found 10-inch crack (including a 2-inch-diameter hole) in aft pressure bulkhead at station 1183. Maintenance repaired per EA 6838.
6	SO039077175	NM	1/18/90	B727	During troubleshooting of a pressurization problem, maintenance found a 12-inch crack on the forward lower corner of the forward lower cargo door. Maintenance replaced forward lower cargo door per MM 52-31-0L, page 406. Pressurized aircraft and no leaks were found. Aircraft good to continue service at this time.
7	GL019077993	WP	12/22/89	DC9	During periodic C-check, found forward door inner pan cracked 12 inches long, and tee fitting broke in half with rivets sheared in beam at upper aft NR 1 beam, and NR 5 beam cracked around inboard latch mechanism. Replaced section of inner pan, replace tee fitting per MM 52-00, SRM 52-05.
8	CE059078122	NM	2/21/90	B747	During OP-16, found NR 3 pylon bathtub fitting cracked approximately 11 inches long around outboard bolt forward side of NAC station 221. Replaced bathtub fitting per SRM 51-10-01. Aircraft total cycles 14,496.

Table 2. Detection Interval Iteration—100 Hour Intervals

Time (Hrs)	Cracks (Inch)	*Detection Probability (%)	Nondetected Probability (%)	Incremented Probability (%)	Accumulating Probability (%)
0	0.05	37	100.0	37.0	37.0
100	0.07	38	63.0	23.9	60.9
200	0.08	38	39.1	14.8	75.7
300	0.09	39	24.3	9.4	85.1
400	0.10	40	14.9	5.9	91.0
500	0.13	41	9.0	3.6	94.6
600	0.16	42	5.4	2.2	96.8
700	0.21	43	3.2	1.3	98.1

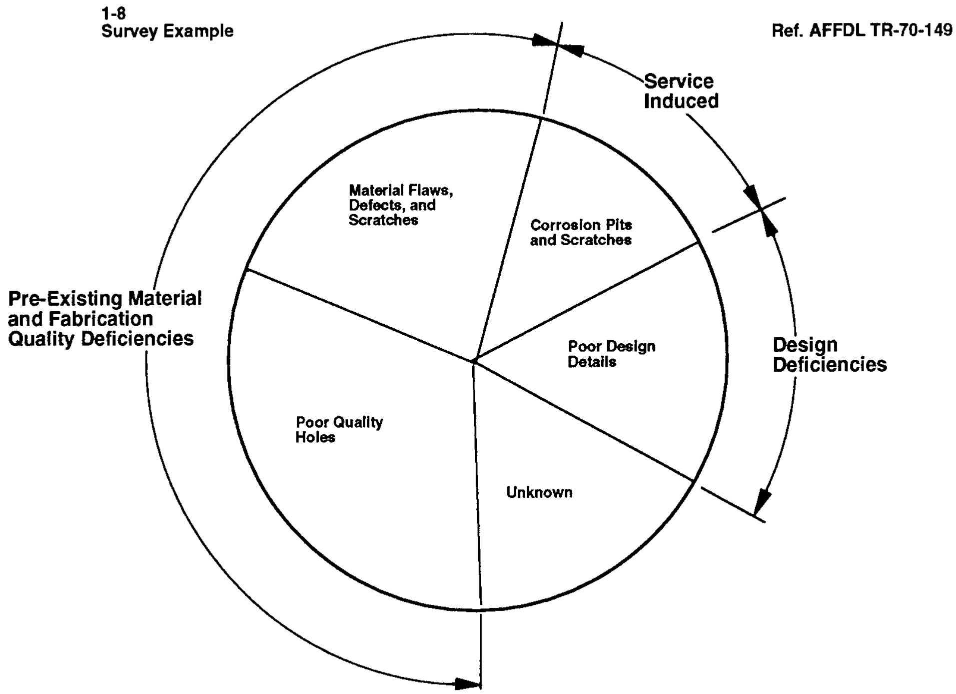
*At a 95% confidence level.

Table 3. Detection Interval Iteration—200 Hour Interval with Detection Probability Increased by 8%

Time (Hrs)	Cracks (Inch)	Detection Probability (%)	Nondetection Probability (%)	Incremental Probability (%)	Accumulating Probability (%)
0	0.05	45	100.0	45.0	45.0
200	0.08	46	55.0	25.3	70.3
400	0.10	48	29.7	14.3	84.6
600	0.16	50	15.4	7.7	92.3

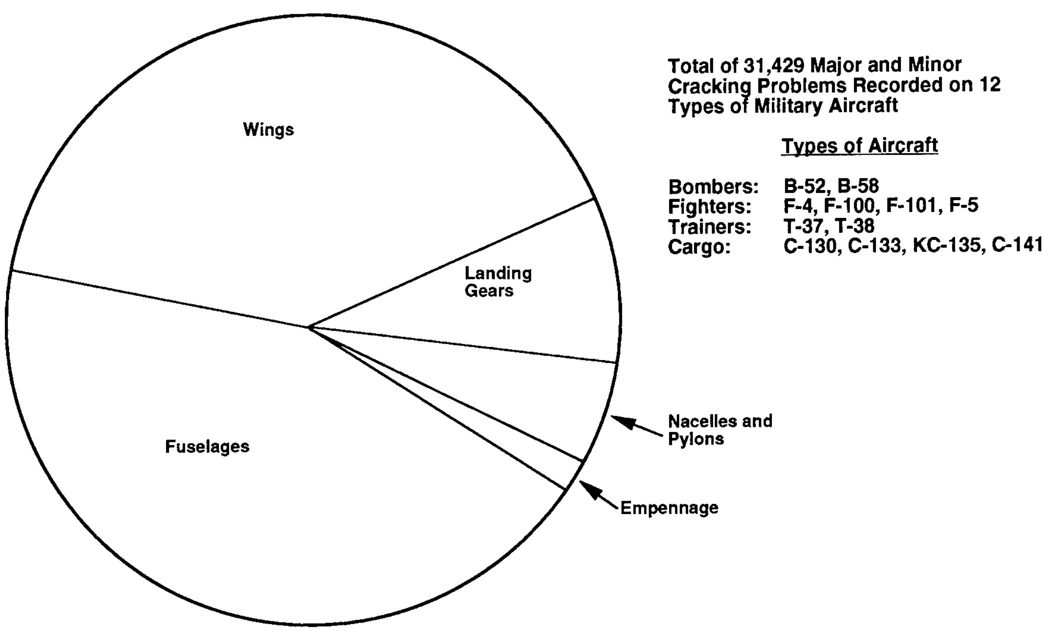
Table 4. Summary of Effects of Conditions on Each Inspector

Factors which may affect performance	Inspector								Totals
	4	5	6	7	8	9	10	11	
Boring work	4	1	4	3	4	1	1	5	23
Temperature	4	1	1	4	1	3	4	4	22
Sleeping difficulties	2	2	2	3	5	1	2	5	22
Change of shift	3	2	4	1	1	2	1	3	17
High workload	1	2	3	1	4	1	1	4	17
Being away from home	3	2	1	1	1	1	2	4	15
Being alone	2	1	1	1	1	1	1	4	12
Noise	1	1	1	1	4	1	1	1	11
The cameras	2	1	1	1	1	1	2	1	10
Totals	22	13	18	16	22	12	15	31	



Based on a study of 64 major cracking/failure problems on military aircraft (6 month period)

Figure 1. Cracking and Failure Origins



Survey of AFLC AMF 66-1 Data Bank (Aug 1968 through Jan 1969) Ref. AFFDL TR-70-149

Figure 2. Examples of Distribution and Magnitude of Service Cracking Problems

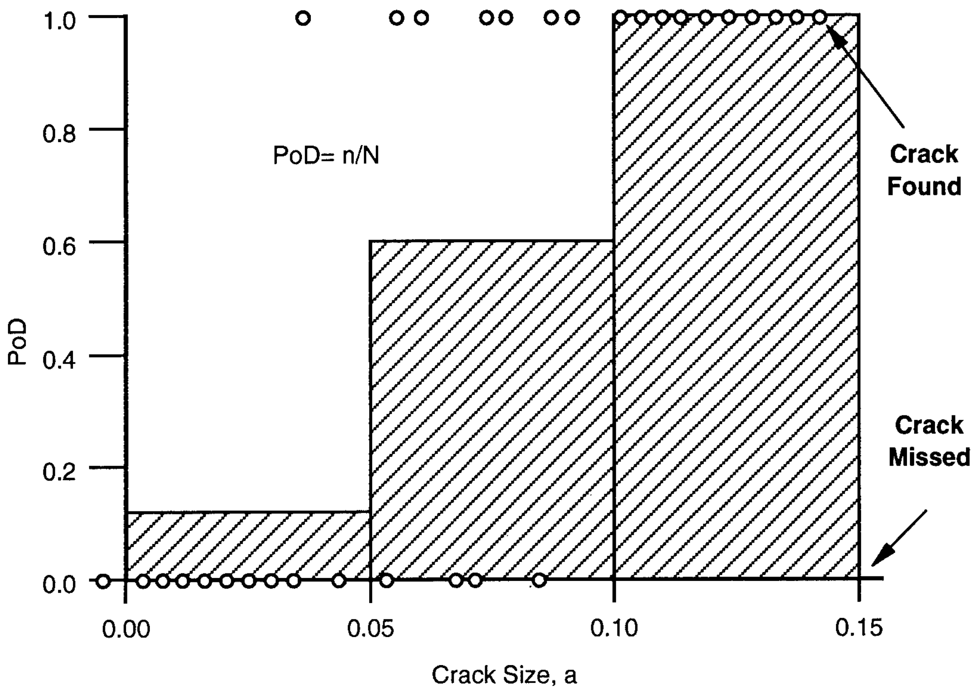


Figure 3. Resolution in PoD vs. Resolution in Crack Size

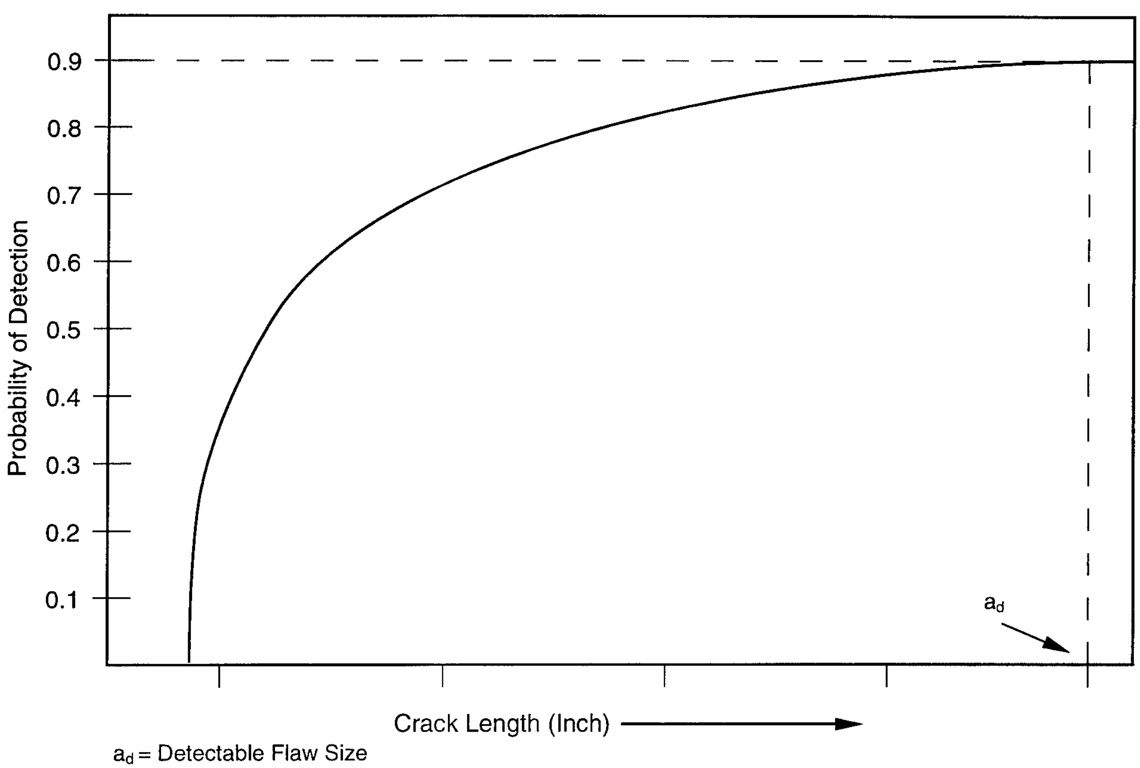


Figure 4. Typical Probability of Detection Curve

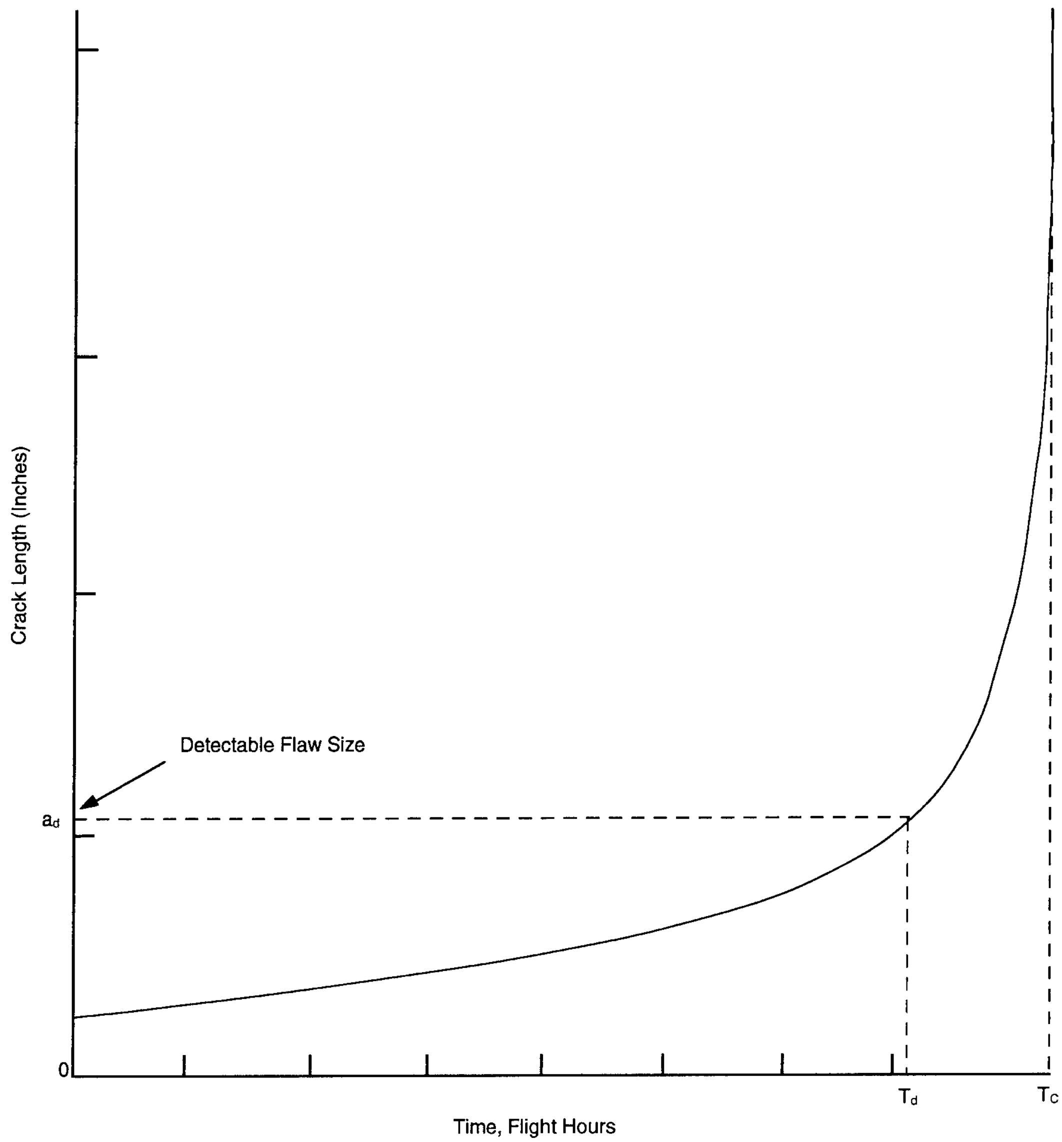


Figure 5. Canadair Calculated Crack Growth Curve

No Crack Present, None Indicated	**Crack Present, None Indicated**
No Crack Present, Crack Indicated	**Crack Present, Crack Indicated**

Figure 6. Possible Outcomes of Inspection Tasks

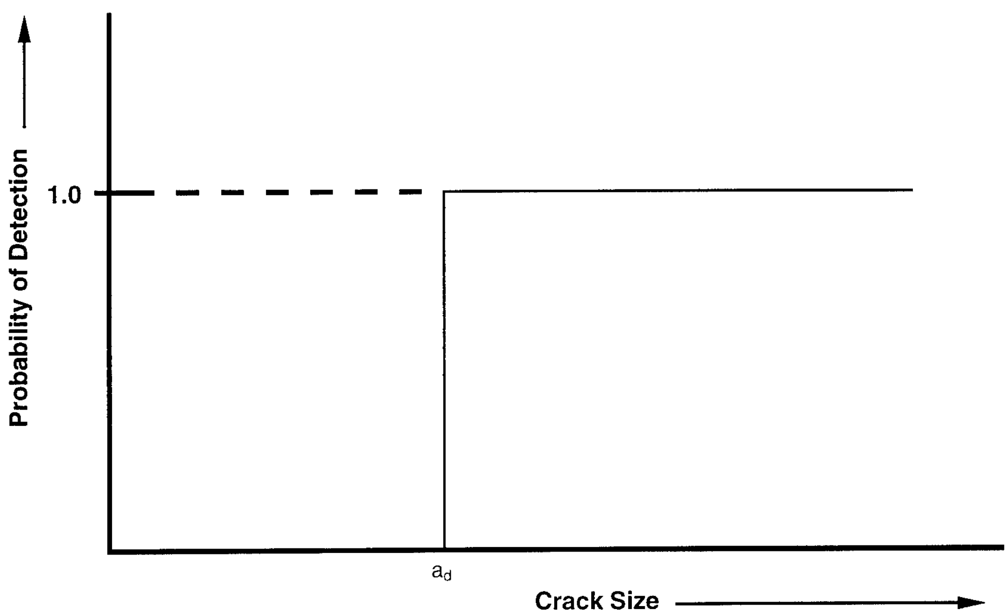


Figure 7. Ideal PoD Curve to Eliminate False Calls

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14. Abstract

This Lecture Series covers systems update and structural airworthiness aspects of aging, fixed-wing aircraft. It highlights the aspect of retrofit/rejuvenation of aging aircraft through presentations relating to three front-line combat aircraft in NATO's inventory. The presentations describe implementation strategies and ways to improve the ability of an airframe to accommodate new systems to meet present day mission requirements. Technical issues pertaining to structural safety, maintenance management, and proactive rehabilitation/retrofit schemes are also discussed.

It provides technical information to fleet operators and managers to assist them to better manage aging aircraft fleets and be able to deal with aging related problems as they arise. It also targets industry personnel responsible for upgrading the capabilities of combat aircraft, maintenance personnel at air logistics centers, and specialists involved with the design of repairs and prescription of inspection methods.

"Aging aircraft" has several connotations, amongst them technological obsolescence, performance upgrading, changing mission requirements unanticipated during design specification and development, the specter of runaway maintenance costs, decreased safety, impairment of fleet readiness, and unavailability of home depot facilities.

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